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Lunar Reconnaissance Orbiter Project

Electrical Systems Specification

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Goddard Space Flight Center
Greenbelt, Maryland

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CM FOREWORD

This document is a Lunar Reconnaissance Orbiter (LRO) Configuration Management (CM)-controlled document. Changes to this document require prior approval of the applicable Configuration Control Board (CCB) Chairperson or designee. Proposed changes shall be submitted to the LRO CM Office (CMO), along with supportive material justifying the proposed change. Changes to this document will be made by complete revision.

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LUNAR RECONNAISSANCE ORBITER PROJECT

DOCUMENT CHANGE RECORD

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REV LEVEL	DESCRIPTION OF CHANGE	APPROVED BY	DATE APPROVED
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List of TBDs/TBRs

Item No.	Location	Summary	Ind./Org.	Due Date
1	Section 3.1.2.7	Modeling method simulating power bus impedance.	P. Luers/ GSFC	8/1/2005
2	Section 3.3.2 d)	Frequency of LRO S-Band uplink (receiver center frequency) for RE02 testing	J. Soloff / GSFC	12/31/2005
3	Section 3.3.4, Table 3-2	Frequency of LRO S-band downlink (S-band transmitter center frequency) for RS03 testing	J. Soloff / GSFC	12/31/2005
4	Table 5-1	Verification Matrix Table	P. Luers / GSFC	8/1/2005
5	Appendix B	Traceability Matrix	P. Luers / GSFC	8/1/2005

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1.0 INTRODUCTION

1.1 PURPOSE

This document provides the electrical and electronic requirements and some guidelines to the subsystem designers for the Lunar Reconnaissance Orbiter (LRO) mission. The purpose of these requirements and guidelines is to assure reliable and compatible operation of the elements that make up the electrical system, both during ground testing and on orbit, with margin for both expected and worst-case environmental conditions.

Specific details of each subsystem interface will be documented in subsystem specifications, component specifications, and interface control documents (ICDs).

1.2 DEFINITIONS

Component: A functional subdivision of a subsystem and generally a self-contained combination of items performing a function necessary for the subsystem's operation. Examples are electronic box, transmitter, gyro package, actuator, motor, battery.

Instrument: A spacecraft (SC) subsystem consisting of sensors and associated hardware for making measurements or observations in space. For the purposes of this document, an instrument is considered a subsystem (of the SC).

Orbiter: An integrated assemblage of modules, subsystems, etc., designed to perform a specified mission in space. For the purposes of this document, “Orbiter” and "SC" are used interchangeably. Other terms used to designate this level of assembly are Laboratory, Observatory, and satellite.

Subsystem: A functional subdivision of a SC consisting of one or more components. Examples are structural, thermal, attitude control, electrical power, command and data handling (C&DH), communication, science instruments or experiments.

1.3 ELECTRICAL SYSTEM OVERVIEW

The electrical system includes electronics and electrical components, interconnect harnessing, structural chassis grounding system, grounding of external coatings, thermal blankets and elements that provide shielding.

The environments that apply to the electrical system include self-generated, conducted, and radiated electromagnetic noise; ground-based electromagnetic emitters, the thermal and mechanical environments of integration and test (I&T), the launch environment, and the on-orbit environment.

1.3.1 Electrical System

The LRO electrical system includes the electrical elements mounted on the Orbiter that are interconnected to perform their defined functions to meet mission requirements. To the extent that there are challenges in interconnecting the electrical system, whether built in-house or procured from an external vendor, this specification is designed to define those design aspects that are critical to the integrated functioning of the system. This ICD defines the LRO general and specific electrical requirements. The LRO subsystems should implement these requirements during their design process in order to assure proper system operation and will verify that the requirements are met.

1.3.2 Electrical System Drivers for LRO

The electrical system must be designed carefully to address electromagnetic interference (EMI), the orbital charging and radiation environments, as well as the functional requirements of collecting instrument data and transmitting it to Earth.

a. On-Orbit Charging Environment: While electrostatic discharge (ESD) threats may not be totally eliminated, they can be minimized and their effects mitigated through the use of sound design practices. An overview of the planned LRO integrated approach (Section 3.7, Charging and Discharging Requirements) includes limiting the number of discharge sources, limiting the size of discharges, implementing shielding between potential sources and potential victims of discharges, and controlling victim susceptibility via filtering and bandwidth control.

b. EMI: The key element of EMI control is the design and use of the SC structure as a Faraday cage. The Faraday cage concept provides shielding between the noisy outside environment and the electronics and harnessing internal to the SC. Harnesses that transition through the Faraday cage will be grounded, shielded, and/or filtered to maintain the overall shield integrity. Noise sources external to the SC are expected to include unavoidable ESDs, ground-based Radio Frequency (RF) emitters, and self-generated RF from the LRO Ka- and S-band RF systems.

c. Instrument Suite: The LRO instrument suite, as well as Star Trackers, will contain instruments with Charge-Coupled Device (CCD) detectors. These detectors are sensitive to common mode noise and can pick up ground noise.

   To minimize the total common mode noise environment, noise sources will be controlled at the potential sources by limiting alternating current (AC) noise (Section 3.2, System Grounding Requirements). Coupling mechanisms between the potential sources and the CCD victims will be controlled by providing a low AC impedance to chassis ground.

d. High Data Rate: The LRO downlink data rate of 125 megabits per second (Mbps) requires relatively high-frequency clocks with corresponding fast rise and fall times that may represent a significant noise source. High data quality and integrity requirements, along with short bit times for the telemetry data, represent challenges to maintaining error-free data. These high-frequency clocks are basically RF signals and need to be treated as such when it comes to both the grounding approach and interfacing via impedance matched transmission lines.

e. Total Dose Radiation and Single-Event Effects (SEEs): The LRO orbit represents a moderate radiation environment. The planned approach for total dose shielding will be shared between the structure and the component chassis to limit the radiation at the part level.

   SEEs will be controlled through the use of radiation-hard or radiation-tolerant parts and circuit designs that can tolerate Single-Event Upsets (SEUs). Potentially destructive damage will be controlled through the use of radiation-hardened parts, while upsets or soft failures will be controlled through radiation-tolerant parts, circuit design, software design or other mitigation methods. Refer to the Radiation Requirements for the Lunar Reconnaissance Orbiter (431-RQMT-000045) for additional details.

f. Compatible and robust interfaces between electrical system components are key to meeting requirements given the potential noise sources described in the above paragraphs a-d. Special attention should be given to the design and control interfaces. Most low data rate signal interfaces are expected to use the 1553 data bus, which is inherently robust. Non-1553 interfaces will be carefully controlled and reviewed for interface robustness and compatibility, as well as evaluating the potential of being a noise source or noise susceptible victim. In order to minimize the potential for noise problems, each interface will be expected to control its signal bandwidth only to that which is necessary to perform its function, subject to review on a case-by-case basis by the LRO Project. This is expected to encompass rise/fall time control of edges on transmission signals, as well as filtering at the receiving end.

g. RF Environment: The LRO RF environment will include self-generated S-band and Ka-band radiation emitted by the High-Gain Antenna (HGA). At certain pointing angles, this potential EMI source may illuminate the instruments and the solar array (SA). Other self-generated RF sources are expected to include the S-band downlink via the omni antennas. Ground-based RF sources are expected to include the launch pad transmitters and ascent sources, including the ground radars and the launch vehicle. The flight environment is also expected to include ground radars and uplink sources to other SC in the vicinity of LRO.

2.0 DOCUMENTATION

2.1 APPLICABLE DOCUMENTS

2.1.1 NASA Documents

GSFC-STD-7000	General Environmental Verification Standard for GSFC Flight Programs and Projects (GEVS)

2.1.2 Non-NASA Documents

MIL-STD-461C	EMI/EMC Requirements Electromagnetic Emission and Susceptibility Requirements for the Control of Electromagnetic Interference
MIL-STD-462	EMI/EMC Testing Methods, Notice 6
ECSS-E-50-12	ESA SpaceWire Specification
TIA/EIA-422	Electrical Characteristics of Balanced Voltage Digital Interface Circuits (formerly known as RS-422)
TIA/EIA-644	Electrical Characteristics of Low Voltage Differential Signaling (LVDS) Interface Circuits

2.2 REFERENCE DOCUMENTS

2.2.1 NASA Documents

431-SPEC-000222	Lunar Reconnaissance Orbiter Project Power Distribution Diagram Specification
431-SPEC-000103	LRO SpaceWire Specification
431-SPEC-000091	Thermal Systems Specification
EEE-INST-002	Instructions for EEE Parts Selection, Screening, Qualification, and Derating
431-SPEC-000013	Lunar Reconnaissance Orbiter Project Power Subsystem Electronics Specification
565-PG-8700.2.1	Design and Development Guidelines for Spaceflight Electrical Harnesses
NASA-HDBK-4002	Avoiding Problems Caused by Spacecraft On-Orbit Internal Charging Effects
NASA-HDBK-4001	Electrical Grounding Architecture for Unmanned Spacecraft
SEECA	Single Event Effect Criticality Analysis <http://radhome.gsfc.nasa.gov/radhome/papers/seecai.htm>
TP2361	Design Guidelines for Assessing and Controlling Spacecraft Charging Effects

2.2.2 Non-NASA Documents

AFSCM 91-710	Range Safety User Requirements Manual
MDC 00H0016	Delta II Payload Planners Guide
MIL-HDBK-1553	Multiplexed Data Bus Handbook
MIL-STD-1553B	Multiplexed Data Bus
MIL-STD-1576	Electroexplosive Subsystem Safety Requirements and Test Methods for Space Systems

3.0 ELECTRICAL SYSTEM REQUIREMENTS

In this document, a requirement is identified by "shall," a good practice by "should", permission by "may", or "can", expectation by "will", and descriptive material by "is."

3.1 POWER

ESS-1: The power subsystem shall supply switched and unswitched unregulated power services with the specifications below.

3.1.1 Power Distribution and Switching Scheme

ESS-2: Distributed power architecture shall be used to supply over-current protected power to all the loads. The power subsystem will provide +28 volts (V) unregulated power to each subsystem.

ESS-3: No +28V power returns shall be switched.

Unswitched power will be supplied to only those critical functions necessary to receive commands and manage redundancy from the ground. Except for the previously mentioned critical functions, all other functions will be commandable to the off state in order to turn off a failed load or to conserve power in an emergency. The Lunar Robotic Orbiter Power Distribution Diagram Specification (431-SPEC-000222) shows a simplified LRO power distribution diagram with default power bus switch states.

All power services will be over-current protected (based on service capacity) using solid-state power controllers (SSPCs). All SSPCs will act as circuit breakers, tripping off at preset current levels, and they are designed to be re-settable by command. See Section 3.1.3.5 b) for derating.

ESS-4: Unswitched power services shall use fuses as over-current protection devices during integration and test only.

The current sensing shunts for telemetry are located in the return portion of the switched and unswitched services. Current must be returned on the originating service.

3.1.2 Power System Electronics Specifications

For internal Power System Requirements, refer to the Lunar Reconnaissance Orbiter Project Power Subsystem Electronics Specification (431-SPEC-000013) and the Lunar Reconnaissance Orbiter Project Power Distribution Diagram Specification (431-SPEC-000222).

3.1.2.1 LRO Power Redundancy

ESS-5: The power system shall provide the capability for redundant wires (more than required for current carrying for each service.

3.1.2.2 Power System Electronics Output Switching Profile

ESS-6: When a service is switched on, the voltage shall rise from 0 to the steady-state voltage no faster than 50 microseconds (μs) to reduce the in-rush at the user circuitry.

ESS-7: When a service is switched on, the voltage shall rise from 0 to the steady-state voltage no slower than 3 milliseconds (ms) to allow for proper operation of power-on reset circuitry.

ESS-8: When a service is switched off or trips off due to a fault condition, the voltage shall fall to 0 V no faster than 20 μs, prohibiting a sharp turn-off from producing an induced EMI emission.

3.1.2.3 Power Steady State Power

ESS-9: The bus voltage at the Power System Electronics (PSE) Output Module shall have a nominal +31 volts direct current (VDC) output voltage with a range from +21 VDC to +35 VDC (inclusive) at the component end of the electrical harness.

ESS-10: The peak current shall not exceed 80% of the SSPC maximum sustainable current, shown below in Table 3-1.

Table 3-1. Switched Services Currents and Deratings

Switched Service Type (A)	SSPC Maximum Sustainable Current (A)	80% Derating (A)
1	1.2	0.96
2	2.4	1.9
5	6	4.8
10	12	9.6
15	18	14.4

3.1.2.4 Power Bus Ripple

ESS-11: The bus ripple contributed by PSE shall be less than 0.3V peak-to-peak (p-p).

ESS-12: Nominal orbiter level power bus ripple resulting from contributions from all nominal sources shall be less than 1.0V p-p over the frequency range of 1.0 hertz (Hz) to 10 megahertz (MHz), and 0.5V p-p over 10 MHz at the power system outputs, under any load condition.

3.1.2.5 Single-Event Power Bus Transients

ESS-13: Single-event power bus transients superimposed on the power buses due to normal subsystem load switching shall be limited to +/-3.0V from the steady-state bus value. An example of this would be a subsystem that controls its own loads, turning them on and off.

The bus will recover to within 10% of its steady state value in 10 ms for a positive or negative load step of 10 amps with a maximum current rate change of 50 milliamps per microsecond (mA/µs). The bus will recover to within 10% of its steady state value in 50 ms for a positive or negative load step of 15 amps with a maximum current rate change of 300 mA/µs.

3.1.2.6 Over-Current Protection Deratings

ESS-14: SSPC devices used for LRO shall be selected based on the load current for each load, and derated per the solid-state power switch device I^2T trip curve characteristics, which can be found in the parts specification.

3.1.2.7 Bus Impedance

The impedance of the LRO power bus is a combination of the contributing impedance effects of the power source, distribution harness, switching devices, and connectors. For the purposes of modeling, the power subsystem impedance is approximated as an 80-milliohm resistor in series with 2 micro Henrys of inductance on each power and return line. The impedance of the power distribution harness must be added to this model to approximate the impedance of the power bus as seen at any given component power input. This impedance is component specific but may be approximated for test by **TBD** meters of wire, American Wire Gauge (AWG) 22, twisted, non-shielded.

3.1.2.8 Insulation

ESS-15: Exposed battery terminals shall be conformally coated to reduce likelihood of accidental short circuits to the SC structure.

3.1.3 <u>User (Subsystem) Specifications</u>

The specifications in this section apply to all subsystems whether built in-house or procured from an external source.

3.1.3.1 Steady-State Voltage

ESS-16: SC subsystems shall operate in the presence of a +21 VDC to +35 VDC power input at their primary power inputs. The nominal power input at the subsystems will be +31 VDC.

3.1.3.2 Ripple

ESS-17: SC subsystems shall meet operational performance requirements in the presence of ripple as described in Conducted Susceptibility (CS) 01/CS02 testing in Section 3.3.3.1.

3.1.3.3 Turn-on Transients (In-Rush Current)

The SC PSE utilizes SSPC devices to control the power. Unlike the electromechanical switches, the solid-state power switch devices control the turn-on time by limiting the input voltage rise time. The typical turn-on time of a solid-state power switch device is between 50 and 200 µs. The input voltage rises linearly with respect to the turn-on time. This delay may eliminate the need for a subsystem to employ an active means for reducing in-rush current at their power input. The subsystem should plan, analyze and test to verify that a current limiter is not required.

ESS-18: The LRO component transient in-rush current shall be within the following limits as listed below and as provided in Figure 3-1 for 1 Amp Services, or Figure 3-2 for 5, 10 and 15 Amp Services.

Note: For a 2 Amp services, refer to Figure 3-1 and multiply the voltage axis by 2.

ESS-19: If the inrush current is measured with an electromechanical device as the input power switching device, the inrush current of a subsystem shall not to exceed a rate of change of 1 amp per microsecond (A/µs) in the first 10 µs.

ESS-20: If the inrush current is measured with an electromechanical device as the input power switching device, the inrush current of a subsystem shall not have a maximum rate of change of greater than or equal to 20 mA/µs after the initial 10 µs surge.

ESS-21: If the inrush current is measured with an electromechanical device as the input power switching device, the subsystem transient current shall never exceed 300% of the maximum steady-state current in the first 10 ms.

ESS-235: If the inrush current is measured with an SSPC as the input power switching device, the inrush current of a subsystem shall not exceed a rate of change of 200 milliamp per microsecond (mA/µs) until the voltage reaches the nominal level.

ESS-236: If the inrush current is measured with an SSPC as the input power switching device, the inrush current of a subsystem shall not exceed a rate of change of 20 mA/µs after the voltage reaches the nominal level.

ESS-237: If the inrush current is measured with an SSPC as the input power switching device, the transient current shall never exceed 300% of the rated output current of the SSPC in the first 70 ms.

ESS-22: In-rush current shall be reduced to nominal load at 100 ms after turn-on.

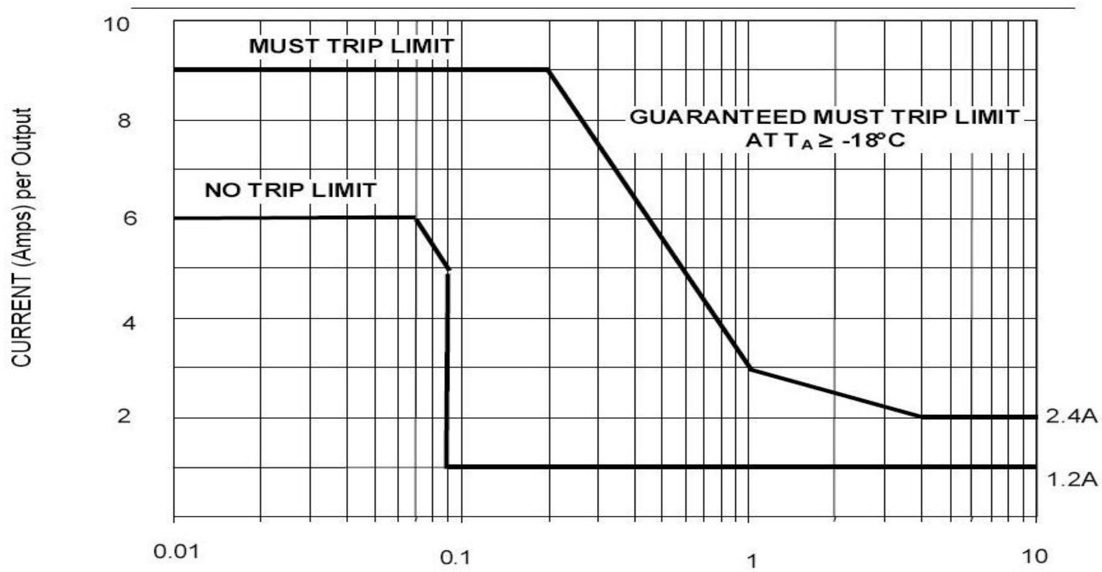


Figure 3-1. SSPC In-rush and Trip Current Limits Curve, 1 Amp Service

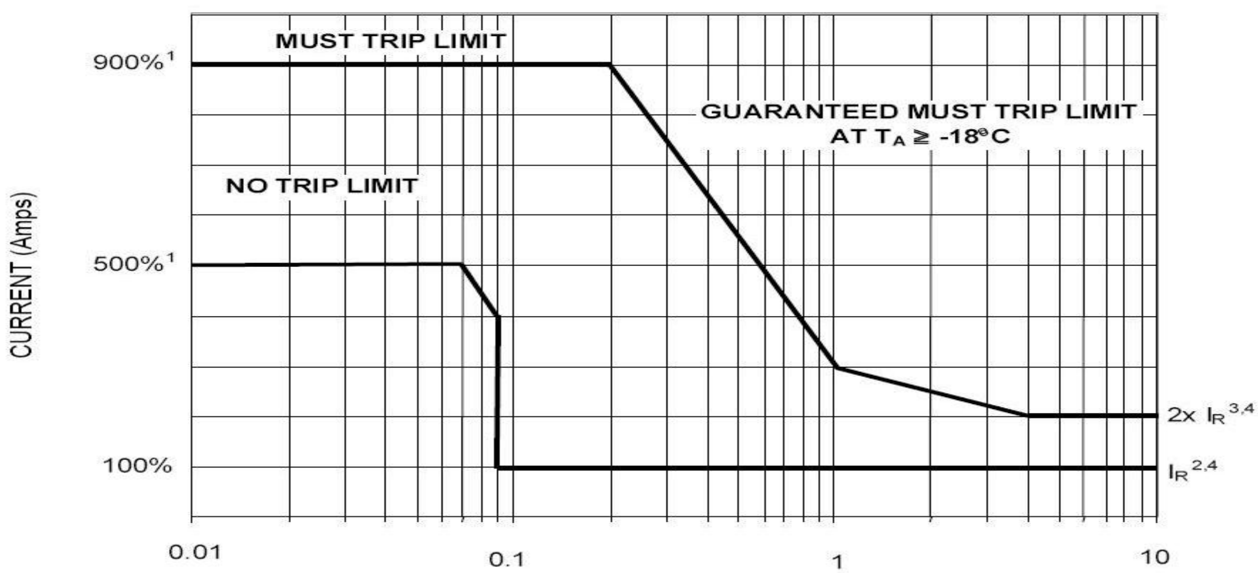


Figure 3-2. SSPC In-rush and Trip Current Limits Curve, 5, 10, and 15 Amp Services

3.1.3.4 Survival of Anomalous Voltage

ESS-23: All subsystems shall be designed to not be damaged by any voltage in the range of 0 to +40 VDC for an indefinite time period applied to the power input during anomalistic operations.

No flight component will be subjected to these tests.

ESS-24: Verification shall be by analysis or test on an engineering test unit (ETU) or at a board level only.

ESS-25: If any subsystem includes components that are not guaranteed to perform down to 0 VDC, anomalous voltage analysis or test shall low limit shall be:

a. lowest voltage guaranteed by manufacturer of that component <u>or</u>

b. low voltage that corresponds to the maximum sustainable current for that subsystem's switched service, whichever is lower

ESS-26: All subsystems shall meet performance requirements during the single-event transient (SET) specified in Section 3.1.2.5.

3.1.3.5 Operational Bus Transients

ESS-27: The rate of change of any operational current transients shall not exceed 20 mA/µs.

3.1.3.6 Turn-Off Transients

ESS-28: When the power service is switched off, the peak voltage transients induced on the power service shall not exceed +40V, nor fall below -1V.

3.1.3.7 Turn-Off Protection

ESS-29: No subsystem shall be damaged by the unannounced removal of power.

ESS-30: Any operations required on a routine basis prior to power turn off shall be listed in that individual subsystem ICD.

ESS-31: Any minimum time following power turn-off that the component must remain off prior to power turn-on shall be listed in that individual subsystem ICD.

3.1.3.8 Redundant Power Supplies

ESS-32: Any subsystem or component that includes redundant power supply inputs shall not be damaged by the simultaneous application of power to both interfaces.

3.1.3.9 Polarity Reversal Protection

ESS-33: All subsystems shall not be damaged by polarity reversal of the input power.

3.1.3.10 Subsystem Over-Current Protection

ESS-34: The use of non-resetting over-current protection (i.e., fuses) within the user subsystems shall be prohibited unless a waiver is requested and approved in accordance with the Robotic Lunar Exploration Program Mission Assurance Requirements (430-RQMT-000006).

3.2 SYSTEM GROUNDING REQUIREMENTS

LRO will use proven grounding techniques that have been shown to reduce EMI and conducted noise from within the system. LRO will use a hybrid grounding approach with the primary structure serving as the low impedance zero voltage reference. Stray AC noise currents are encouraged to flow through structure in order to reduce common mode voltages. Primary power has a DC single point ground and AC multipoint grounds to chassis. Secondary power and grounds are multipoint grounds to chassis.

3.2.1 Single-Point Primary Power Ground

ESS-35: LRO shall implement a single-point grounding scheme for the primary power bus.

ESS-36: The Single-Point Ground (SPG) shall be located within the PSE.

ESS-37: All the primary power returns, SA returns, and battery grounds shall be tied together at the SPG and connected to the SC structure (or SC chassis ground).

ESS-38: At the component primary power interfaces, primary power (28 VDC) and primary power returns shall be isolated from the component chassis by greater than or equal to 1 Megohms (Mohms) direct current (DC).

3.2.2 Distributed Signal Ground

ESS-39: LRO shall implement distributed signal grounding scheme for secondary power returns and/or signal grounds.

ESS-40: The secondary return (power, signal, analog, or digital grounds) shall be locally connected to the component chassis with low impedance paths (<= 2.5 milliohms DC per joint) to minimize stray current.

ESS-41: Both secondary power (or signal) inputs and returns shall be isolated from primary power by equal to or greater than 1 Mohms DC.

3.2.3 Common Mode Noise

ESS-42: Common mode noise for the primary and secondary power, as well as digital, analog and signal grounds, shall be less than 100 millivolts (mV) p-p.

ESS-43: The LRO structure shall not be used intentionally to carry DC current or common mode current.

3.2.4 Bonding or Mating

ESS-44: The LRO structure shall be used as a common reference point for all electronics.

ESS-45: All conductors shall be grounded to the SC structure, no floating conductors.

ESS-46: The electrical DC resistance of a mechanical contact between two conductive mating surfaces shall not exceed 5 milliohms DC.

ESS-47: The primary mating method for a component shall be the metal-to-metal contact between component mounting feet (or baseplate) and the LRO structure. Mating (electrically bonding) surfaces should be free from nonconductive finishes and should establish sufficient conductive contact surface area to satisfy electrical grounding requirement (ESS-51).

ESS-48: If a component is to be mounted on a composite or other low conductive material, a grounding strap shall be attached from the component chassis to an Orbiter conductive structure.

ESS-49: The electrical DC resistance of the grounding strap as measured at the component chassis tie-point and the Orbiter structure tie-point shall not exceed 5 milliohms DC.

ESS-50: All ground straps shall have at least a length-to-width ratio of 5 to 1, be made of copper at least 1 mil thick, and still be flexible enough to allow for bending to occur.

ESS-51: The grounding lug location on the component chassis or the tie points in contact with the ground strap shall have a minimum contact area of 80 millimeters squared (mm2). This is satisfied by a 1/4" diameter fastener and NAS1149C04-series washer.

ESS-52: The electrical DC resistance between any two mated electronic components on the LRO structure, measured at the foot of each component, shall not exceed 10 milliohms DC.

ESS-53: Component connectors and backshells shall be electrically mated to the chassis through an electrical resistance not exceeding 5 milliohms DC.

3.2.5 Shield Grounds

ESS-54: Shield grounds associated with cables (twisted shielded pairs, coaxial cables, etc.) shall be terminated to chassis ground per Figures 3-3 and 3-4.

3.2.6 Grounding of External Orbiter Surfaces

ESS-55: All external surfaces shall be grounded. Where this is not possible, it will be identified in the individual component Electrical ICD.

3.2.6.1 Thermal Blankets

ESS-56: All layers of thermal blankets shall be redundantly grounded to the SC structure using a low-resistance method as follows:

- have at least two grounding tabs for blankets less than 1 meter squared (m^2).
- have an additional grounding tab for every square meter greater than 1 m^2

ESS-57: Any point on the blanket shall be less than 1 meter from the nearest grounding point.

ESS-58: Insulating films such as Kapton and other dielectric materials shall be less than 5 mil thick, assembled to minimize surface charge build-up, and grounded to bleed surface charge.

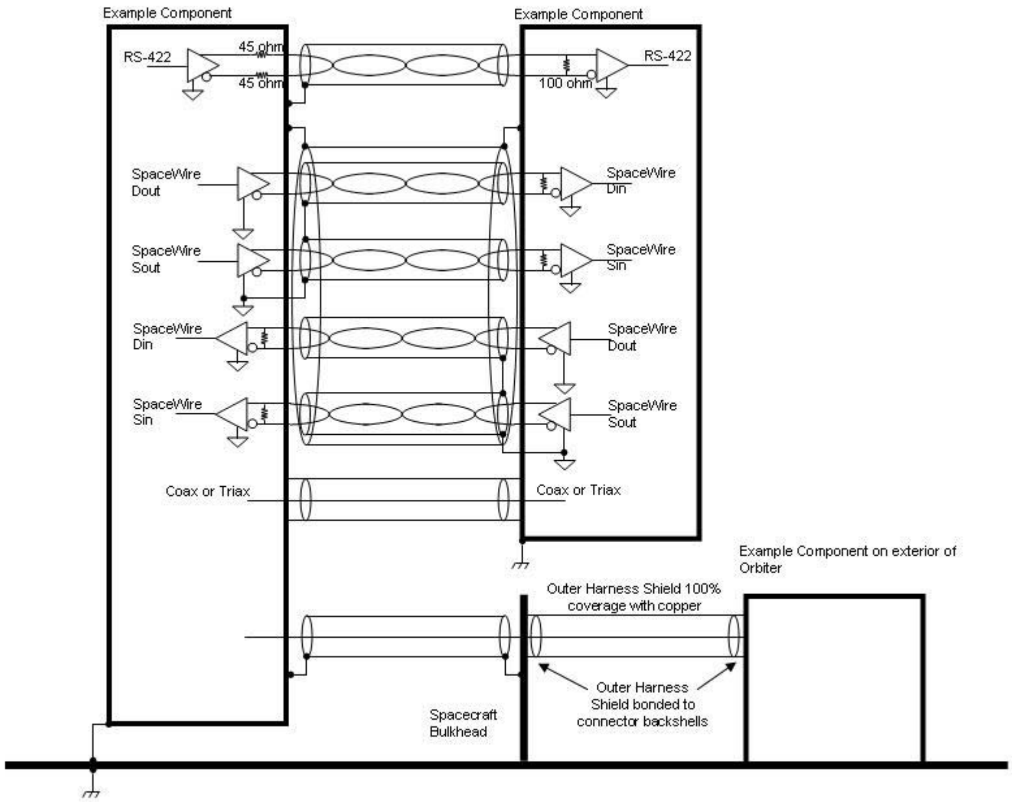


Figure 3-3. LRO Spacecraft Shielding Scheme

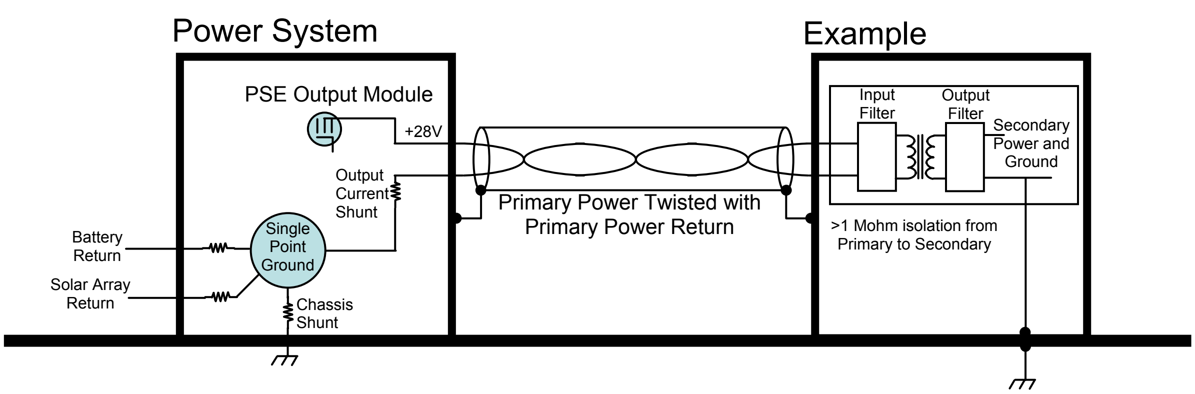


Figure 3-4. LRO Spacecraft Power Isolation Scheme

3.2.6.2 Hinges

ESS-59: A ground strap per Section 3.2.4 shall carry the ground across any hinged joints.

3.2.6.3 Solar Array (SA) Panels

ESS-60: Solar array (SA) panels and substrates shall be electrically grounded to the SC structure.

ESS-61: Ground straps shall be implemented per Section 3.2.4.

3.2.6.4 Antennas and Antenna Booms

ESS-62: The HGA assembly shall employ a grounding scheme to assure HGA metal surfaces and waveguides are grounded directly or indirectly to the SC structure through less than 100 milliohms DC resistance.

The HGA boom gimbal rotating joints and deployment hinges should not be considered adequate in providing a good ground path. Therefore, separate ground connections will be provided in slip-rings on a rotating joint or a dedicated ground strap.

ESS-63: The omnidirectional antenna metal surface and cable shields shall be grounded directly or indirectly to the SC structure through less than 5 milliohms DC resistance.

3.3 EMI/EMC REQUIREMENTS

ESS-64: Emissions and susceptibility testing shall be performed per this document, which has tailored the General Environmental Standards for GSFC Flight Programs and Projects (GEVS) (GSFC-STD-7000) test levels for the LRO mission.

The EMI/Electromagnetic Compatibility (EMC) tests required below are meant to cover the LRO mission environments including Orbiter RF self compatibility, launch site, launch pad, launch/ascent, lunar transfer, and lunar orbit.

ESS-65: Table 3-2 indicates the tests that shall be performed on each component and at the Orbiter level.

Table 3-2. EMI/EMC Applicability and References

	Components (1) (2)	RF Comp (3)	Orbiter	ICD Section	GSFC-STD-7000 Section(4)	461C Section (4)
CE01	X	X		3.3.1.1	2.5.2.1	2
CE03	X	X		3.3.1.1	2.5.2.1	3
CE06		X		3.3.1.2	2.5.2.1	4

	Components (1) (2)	RF Comp (3)	Orbiter	ICD Section	GSFC-STD-7000 Section(4)	461C Section (4)
RE02	X	X	X	3.3.2	2.5.2.2	17
CS01	X	X		3.3.3.1	2.5.3.1a	6
CS02	X	X		3.3.3.2	2.5.3.1a	7
CS03		X		3.3.3.3	2.5.3.1b	8
CS04		X		3.3.3.4	2.5.3.1c	9
CS05		X		3.3.3.5	2.5.3.1d	10
CS06	X	X		3.3.3.6	2.5.3.1e	11
RS03	X	X	X	3.3.4	2.5.3.2	21
Self Compatibility			X	3.3.5		

Notes: (1) X= Applicable
(2) Subsystems may be tested as individual components or assemblies where applicable. Subsystems include instruments (see section 1.2).
(3) GSFC-STD-7000 and MIL-STD-461C sections provided for reference. See applicable ICD section for LRO specific requirements.
(4) RF Components includes all RF receiving components and instruments

Test levels of emissions and susceptibility defined in applicable figures may differ from GSFC-STD-7000 and MIL-STD-461C, this document takes precedence.

ESS-66: The EMI/EMC test methods shall be per the requirements of MIL-STD-462C (Notice 6) unless noted in this document. EMI/EMC requirements will be imposed on individual components.

ESS-67: When fully integrated into the SC, this shall ensure that these components will not interface with each other. In addition, the radiated emission and susceptibility requirements are imposed on a fully integrated SC. This will ensure that the SC will not adversely affect the launch vehicle, will not be affected by the external emissions (particularly at the launch site), and will not interfere with any sensitive instruments making science measurements.

ESS-68: All tests shall be performed in ambient with either the component or system in its most sensitive mode for susceptibility testing and in its most noisy mode as appropriate for the EMI emission test.

3.3.1 Conducted Emissions

3.3.1.1 CE01/CE03

ESS-69: Conducted emissions (CE) from components shall not exceed the values shown in Figure 3-5 when subjected to CE01 (20 Hz – 14 kHz) and CE03 (14 kHz – 40 MHz) narrowband testing.

ESS-70: CE01/CE03 shall be performed on all +28V primary power and return lines to each component.

ESS-71: CE01/CE03 shall be performed in differential and common mode.

Conducted emissions testing will be performed only at the subsystem or component levels.

ESS-72: Each component shall meet the transient current pulse limits, both single event (excluding turn-on) and recurring, as specified in Section 3.1.3. Applicable test parameters and limits are as follows for narrowband conducted emissions:

a) Interface lines to be measured are differential mode current lines: +28V inputs, +28V input returns.

b) Interface lines to be measured are common mode current lines: +28V power inputs with return including heater circuits.

c) Differential mode narrowband test limits are 120 decibel microamps (dBuA) (1.0 A rms) from 30 Hz to 450 Hz, then decreasing to 50 dBuA (10mA rms) at 20 KHz, then decreasing to 20 dBuA (10uA rms) at 2 MHz, and then continuing at that level to 50 MHz, as shown in Figure 3-5.

d) Common mode narrowband test limits are 50 dBuA (0.316 mA rms) from 30 Hz to 20 KHz, then decreasing to 20 dBuA (10uA rms) at 2 MHz, and then continuing at that level to 50 MHz, as shown in Figure 3-5.

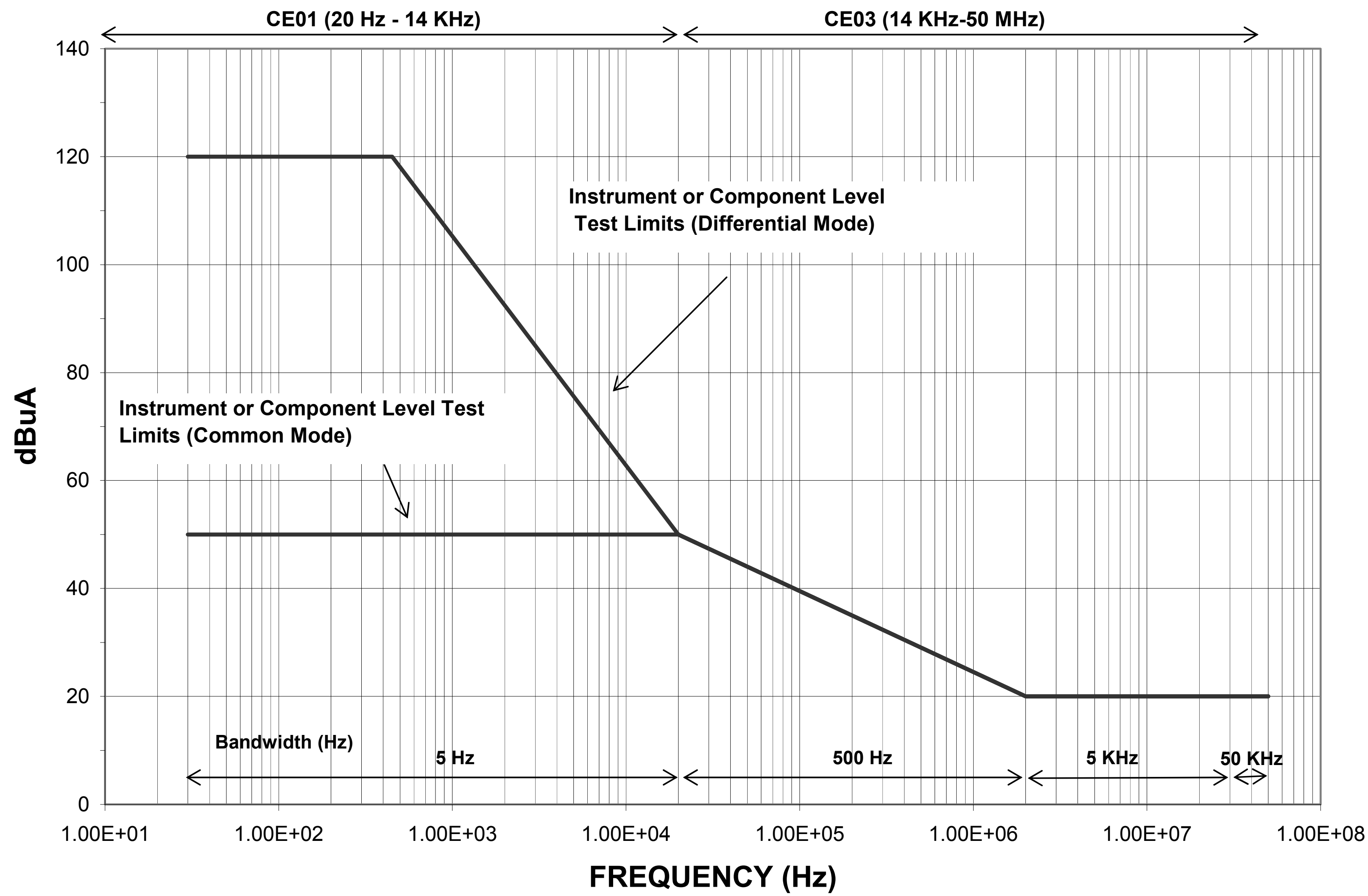


Figure 3-5. Narrowband Conducted Emissions CE01/CE03 Limits

3.3.1.2 CE06

ESS-73: All RF receivers and transmitters shall perform the additional CE06 EMI test to the limits contained in MIL-STD-461C.

3.3.2 Radiated Emissions (RE02)

ESS-74: Radiated emissions (RE) from subsystem or components shall not exceed the values shown in Figure 3-6 or Figure 3-7 when subjected to RE02 narrowband testing.

ESS-75: Radiated electric field emissions from any components that are ON from launch to vehicle separation shall not exceed the limits shown in Figure 3-6 (lower line).

ESS-76: Radiated electric field emissions from any components that are OFF from launch to vehicle separation shall not exceed the limits shown in Figure 3-7.

ESS-77: The aggregate RE from the Orbiter shall not exceed the limits shown in Figure 3-6 (upper line).

The Orbiter receiver has a center frequency at 2100 – 2200 (**TBD**) MHz and the notch in Figure 3-6 will protect the receiver with at least 6 MHz on both sides of the center frequency.

The Delta II maximum allowable payload RE levels are: 38.5 dB microvolt per meter (µV/m) (3-stage) in the 408 - 430 MHz range, as shown in Figure 3-6, and 94.9 dB µV/m (3-stage) at 5.687 - 5.693 Gigahertz (GHz) range, which is not shown in Figure 3-6 (amplitude off the scale).

3.3.3 Conducted Susceptibility

ESS-78: Undesirable response, malfunction, or degradation of performance shall not be produced in any components during CS testing with the tests specified below. Performance deviation of Instruments is acceptable as long as the component under test survives the component CS test. CS testing will not be performed at the Orbiter level.

3.3.3.1 CS01/CS02

ESS-79: The CS01 and CS02 (injection of energy into power lines) shall be performed on all components that contain the DC/DC converters or power regulation devices.

ESS-80: The CS01 test limits for the components level tests shall be 3.1 V rms at the frequency range of 30 Hz to 1.5 kHz, and ramping in a straight line down to 1.0 volt at 50 kHz.

ESS-81: The CS02 limit for the component level test shall be 1.0 V rms at the frequency range of 50 KHz to 400 MHz. These limits, which are defined by MIL-STD-461C, are shown in Figure 3-8 below.

ESS-82: The CS01 and CS02 (injection of energy into power lines) performance shall be verified at the nominal +31V only.

3.3.3.2 CS03

ESS-83: The CS03 (Two Signal Intermodulation) test shall be performed on all RF receiving components.

ESS-84: The CS03 (Two Signal Intermodulation) test performed on all RF receiving equipment shall not cause the RF equipment to exhibit any intermodulation products from two input signals, beyond those permitted in the RF component specification.

ESS-85: The CS03 test for RF receiving components shall be conducted per MILS-STD-462 to the limits specified in GSFC-STD-7000.

3.3.3.3 CS04

ESS-86: The CS04 (Rejection of Undesired Signals) test shall be performed on all RF receiving components.

The CS04 (Rejection of Undesired signals) test for RF receiving components consists of a 0.0 decibel meter (dBm) (1 milliwatt) signal applied directly to the receiver input terminals and notched around the receiver input bandwidth at 80.0 decibels (dB) above its threshold.

ESS-87: The input notch center shall be at the receiver-tuned frequency and in the center of the notch.

ESS-88: The CS04 test for RF receiving components shall be conducted per MIL-STD-461C to the limits specified in GSFC-STD-7000.

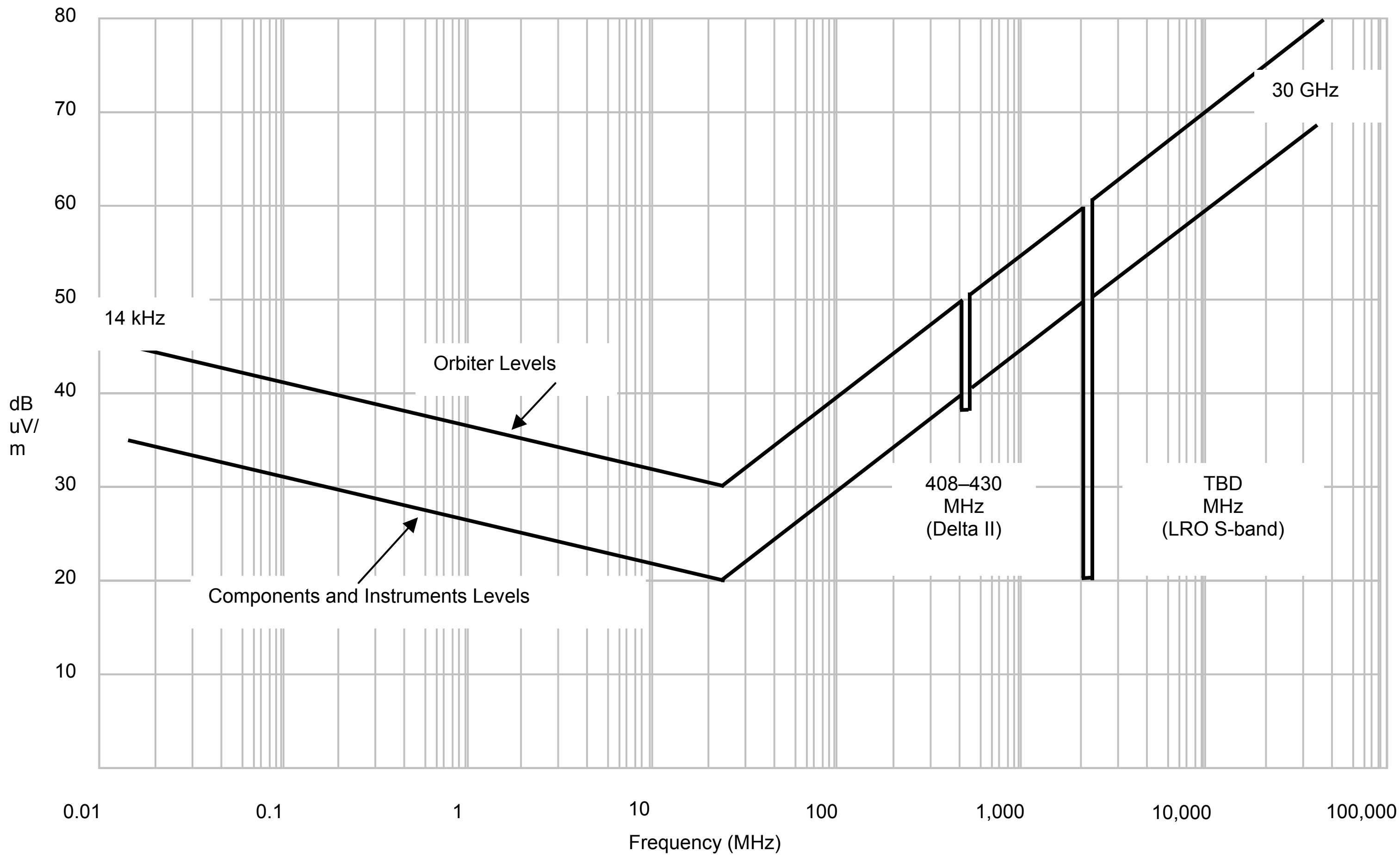


Figure 3-6. RE02 Limits for the Orbiter and Components that are ON from launch to vehicle separation

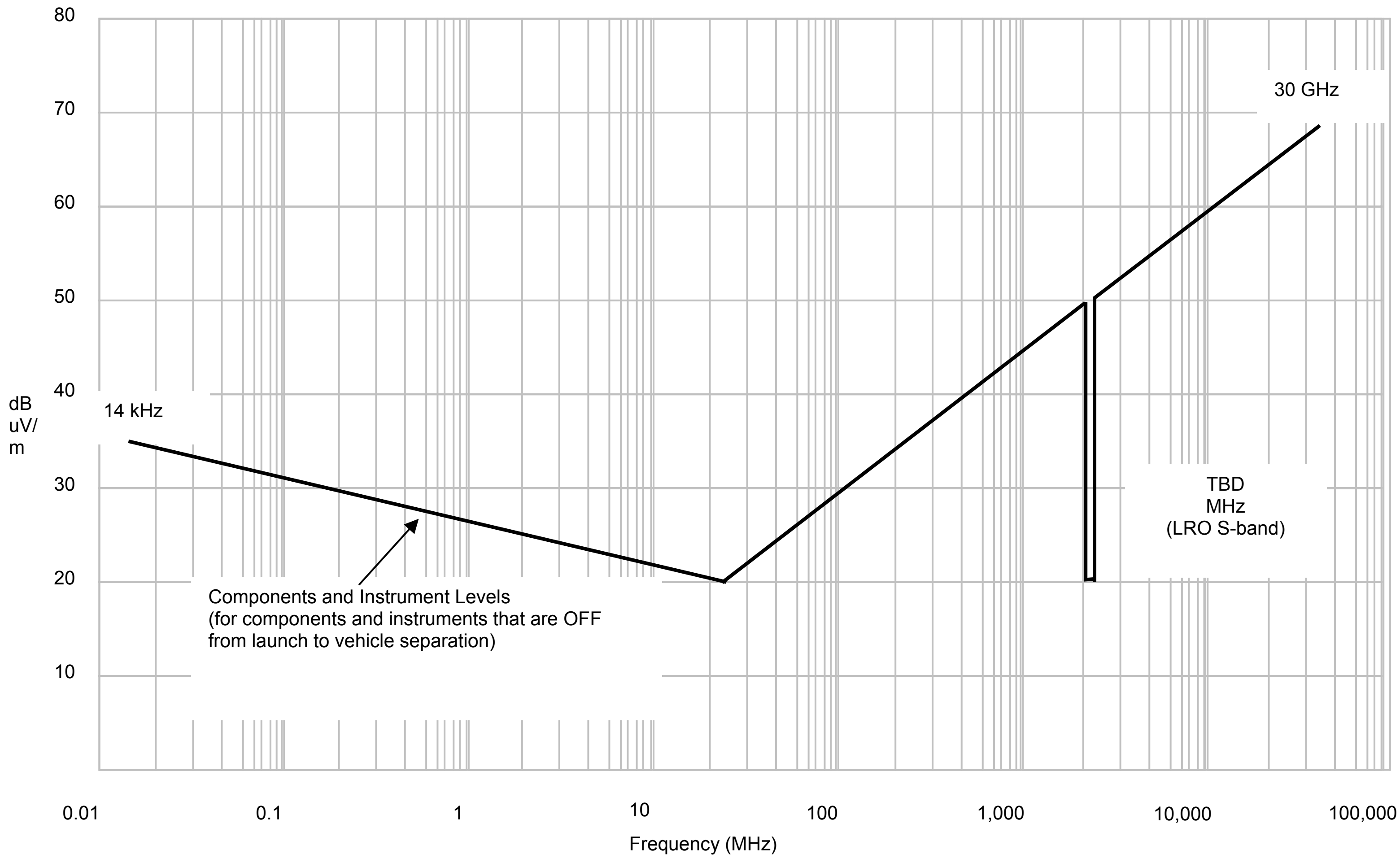


Figure 3-7. RE02 Limits for Components that are OFF from launch to vehicle separation

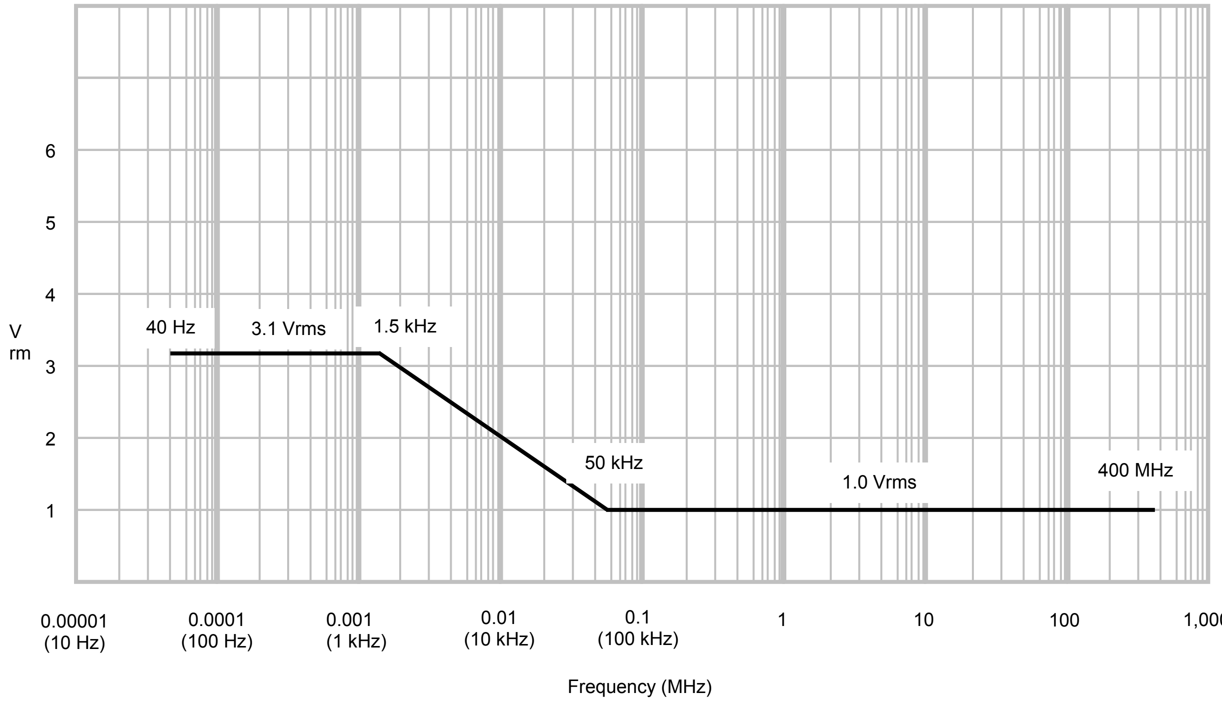


Figure 3-8. CS01/CS02 Limits

3.3.3.4 CS05

ESS-89: The CS05 (Cross Modulation) test shall be performed on all RF receiving components.

ESS-90: The CS05 (cross-modulation) test performed on all RF receiving equipment shall not cause the RF equipment to exhibit any cross-modulation from two input signals.

ESS-91: The CS05 test for RF receiving components shall be conducted per MIL-STD-461C to the limits specified in GSFC-STD-7000.

3.3.3.5 CS06

ESS-92: The CS06 (Powerline Transient) shall be performed on all components that contain the DC/DC converters or power regulation devices.

The CS06 (Powerline Transient) test consists of both a positive transient test and a negative transient test, having amplitude of +28V superimposed on the +28V power bus as shown in Figure 3-9.

ESS-93: This pulse shall be limited to +56V peak absolute value and 10 µs from 0.5E (42V) to the +28V steady-state value crossing point as shown in Figure 3-9.

ESS-94: The CS06 test shall be conducted per MIL-STD-462 to the limits specified in GSFC-STD-7000.

3.3.4 Radiated Susceptibility (RS03)

ESS-95: Undesirable response, malfunction, or degradation of performance shall not be produced during component, or Orbiter Radiated Susceptibility (RS) testing with the E-field levels shown in Table 3-3.

The LRO Expendable Launch Vehicle (ELV) will be serviced and launched from the Cape Canaveral Air Station (CCAS) and can be exposed to the maximum transmitter limits shown in Table 3-4.

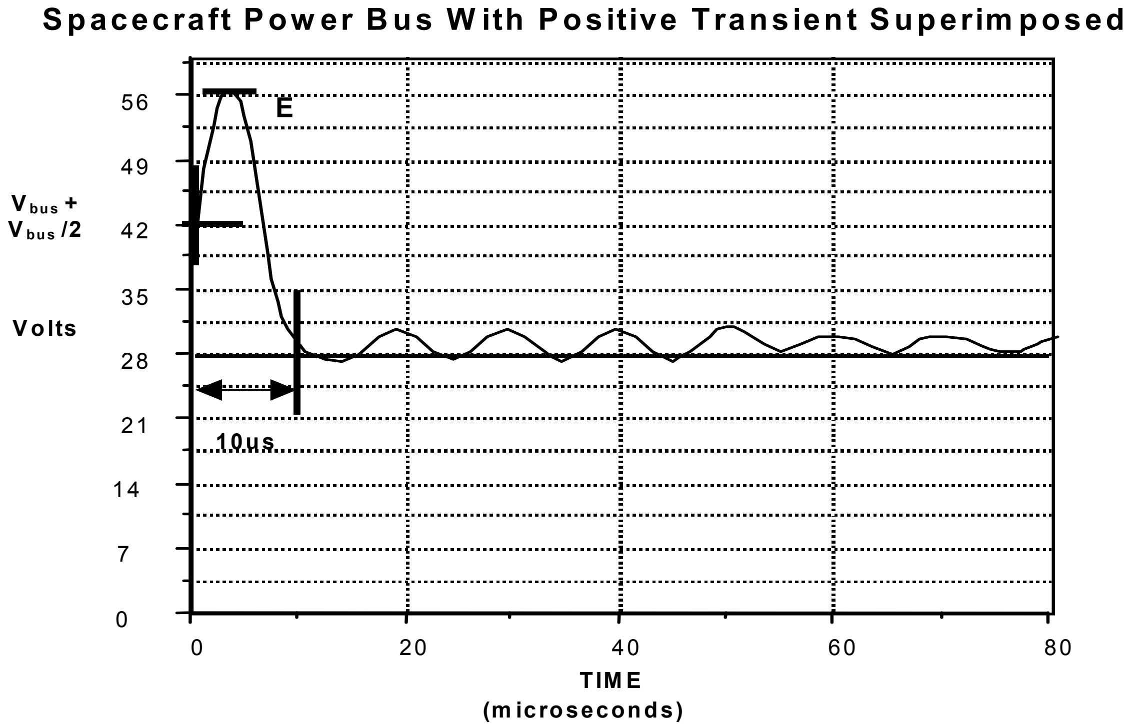
ESS-96: The Orbiter components and instrument shall survive the RS test levels of the launch site transmitters.

Instrument and subsystems that are powered off from launch to launch vehicle separation may be powered off during the RS test at the ELV S-band and C-band transmitter frequencies shown in Table 3-4.

Table 3-3. LRO Operational RS Test Limits

Frequency Range	Test Level	Requirement Source
14 KHz – 2 GHz	2 V/m	GSFC-STD-7000
2 GHz – 12 GHz	5 V/m	GSFC-STD-7000
12 GHz – 28 GHz	10 V/m	GSFC-STD-7000
2.20 (**TBD**) GHz +/- 4 MHz	7 V/m	LRO S-Band Transmitter
25.5 GHz – 28.0 GHz	10 V/m	LRO Ka-Band Indirect Radiation

CS06 Test



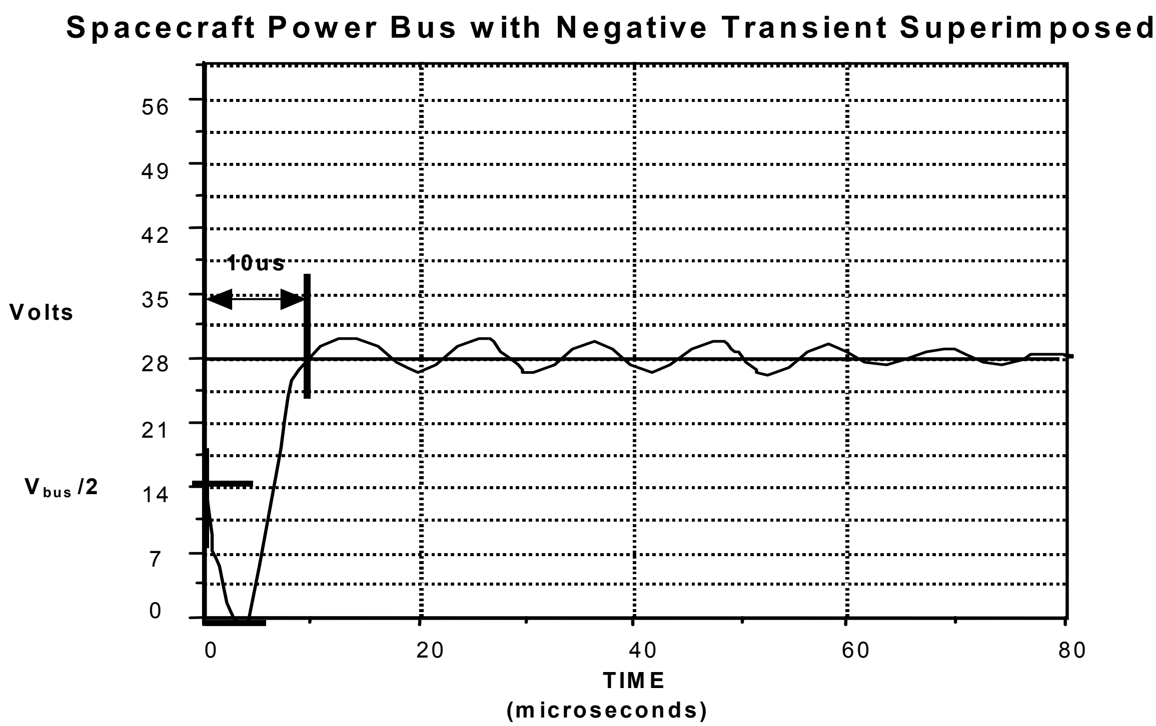


Figure 3-9. CS06 Conducted Susceptibility Test Pulse

Table 3-4. Launch Site/Vehicle RS Test Levels

Frequency Range	Test Level	Requirement Source
14 kHz – 40 GHz	20 V/m	Delta II Launch Pad Environment
2241.5 MHz +/- 650 kHz	40 V/m	Delta II Second Stage S-band T/M
2252.5 MHz +/- 250 kHz	40 V/m	Delta II Third Stage S-band T/M
5765 +/- 6 MHz	40 V/m	Delta II Second Stage C-band beacon (transmit)

3.3.5 Orbiter RF Self-Compatibility

ESS-97: The Orbiter RF self-compatibility test shall be included in the Orbiter-level EMI/EMC test.

ESS-98: During Orbiter Self-Compatibility, the Orbiter shall be configured to a nominal science mode to simulate the in-orbit operation. Ka-band and S-band transmitters will free-radiate from their antennae during this test.

3.4 DATA AND SIGNAL INTERFACES

ESS-99: The presence or absence of any combination of the input signals applied in any sequence shall not cause damage to a component, reduce its life expectancy, or cause any malfunction, whether the component is powered or not.

3.4.1 Inter-Component Communications

ESS-100: All signals between boxes shall be controlled to limit signal bandwidth so that no signal should be given more bandwidth than needed to communicate the necessary functions under all expected on-orbit environmental conditions.

ESS-101: Subsystems connected to the C&DH subsystems via the 1553 data bus shall communicate commands and housekeeping telemetry per Section 3.4.1.1.

ESS-102: Subsystems connected to the C&DH subsystem via the SpaceWire network shall use four twisted pair wires (100+2% ohm cable) with a separate shield around each twisted pair and an overall shield per ESA Space Wire Specification (ECSS-E-50-12).

ESS-103: Subsystems connected using RS-422 differential signals shall adhere to the electrical terminations as given in the Electrical Characteristics of Balanced Voltage Digital Interface Circuits (TIA/EIA-422).

ESS-104: Subsystems connected using LVDS signals shall adhere to the electrical terminations as given in the Electrical Characteristics of Low Voltage Digital Signaling (LVDS) Circuits (TIA/EIA-644).

3.4.1.1 LRO 1553 Data Bus

ESS-105: All 1553 Bus Controller (BC), Remote Terminal (RT), and coupling transformer devices shall comply with MIL-STD-1553B requirements.

3.4.1.1.1 LRO 1553 Data Bus Topology

ESS-106: The transformer-coupled (long stub) interface shall be implemented for the LRO 1553B data bus, as specified in MIL-STD-1553B.

A typical subsystem RT will interface to the data bus with an isolation transformer and a coupling transformer, as shown in Figure 3-10. The complete 1553 data bus will be assembled as shown in Figure 3-11. The LRO-unique 1553B bus implementation requirements and clarifications are described below.

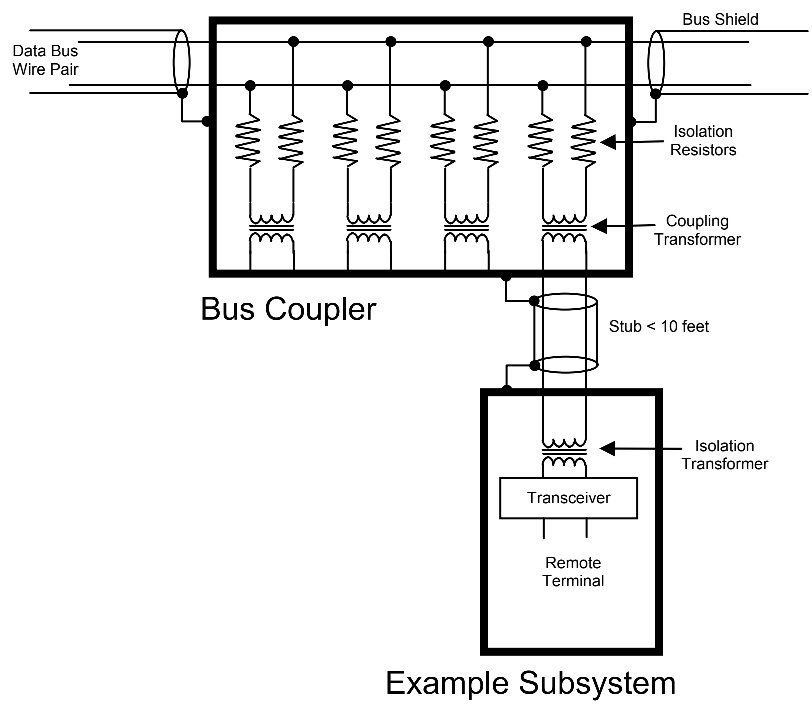


Figure 3-10. 1553 Data Bus Subsystem Diagram

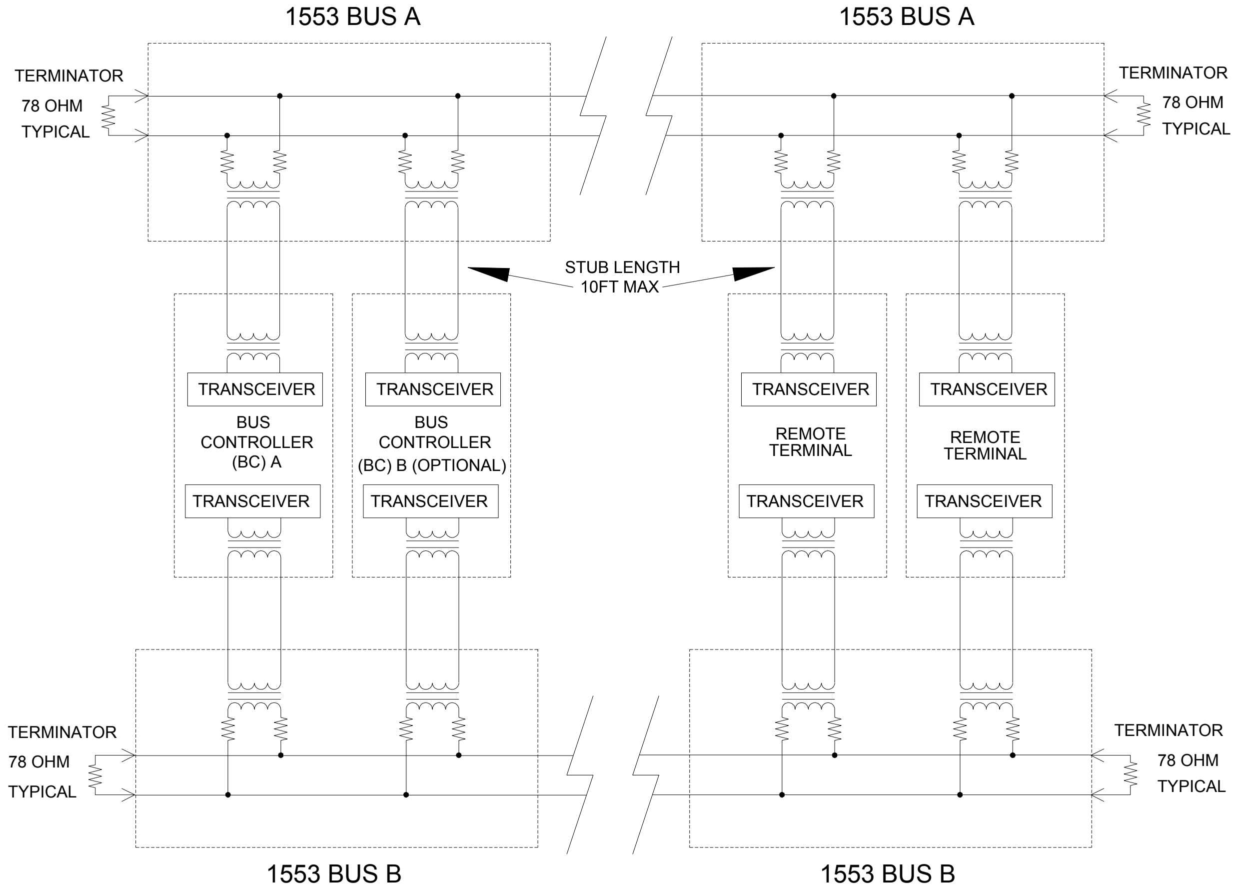


Figure 3-11. 1553 Data Bus System Diagram

3.4.1.1.2 LRO 1553 Remote Terminal Address Assignments

ESS-107: The Remote Terminal Addresses shall be assigned per Table 3-5.

Table 3-5. LRO 1553 Remote Terminal Addresses

Address	Component or Subsystem
0	Reserved (Mode Code)
1	Reserved (Single Board Computer (SBC) #1 RT operations if necessary)
2	Reserved (SBC #2 if necessary)
3	C&DH Thermal Card
4	PSE
5	Spare
6	Spare

Address	Component or Subsystem
7	Spare
8	Integrated Reaction Wheel (IRW) 1
9	IRW 2
10	IRW 3
11	IRW 4
12	Star Tracker (ST) 1
13	ST 2
14	Inertial Measurement Unit (IMU)
15	Reserved (Backup IMU if necessary)
16	CRaTER
17	Diviner Lunar Radiometer Experiment (DLRE)
18	LEND
19	Spare
20	Spare
21	LOLA
22	Spare
23	Spare
24	Propulsion / Deployment Electronics (PDE) A
25	PDE B
26	PDE C
27	PDE D
28	SA Gimbal Control
29	HGA Gimbal Control
30	Spare
31	Reserved (Broadcast)

3.4.1.1.3 LRO 1553 Redundancy

ESS-108: A dual, standby-redundant bus as defined in MIL-STD-1553B shall be used.

3.4.1.1.4 LRO 1553 Data Bus Connectors

ESS-109: All 1553 Bus Controller(s) and Remote Terminals (RTs) shall utilize a connector compatible with the LRO 1553 harness which uses a Trompeter PL3155AC connector.

3.4.1.1.5 LRO 1553 Data Bus Cable

ESS-110: The 1553 data bus shall use M17/176-00002 cable.

3.4.1.1.6 LRO 1553 Stub Coupling

ESS-111: The 1553 bus shall use transformer coupling as defined in Section 4.5.1.5.1 of MIL-STD-1553B.

ESS-112: The 1553 bus coupling transformer shall have a turn ratio of 1:1.41+/-3% (stub to bus).

ESS-113: The 1553 bus coupling transformer shall include fault isolation resistors in accordance with Section 4.5.1.5.1.2 of MIL-STD-1553B with Z_o = 70 ohms +/-10%

ESS-114: The 1553 bus shall have a characteristic impedance of 78 ohms and be terminated at both ends of the bus with 78 ohms termination.

ESS-115: The 1553B bus cabling and couplers shall have 100% shielding coverage.

The 1553 stub should be as short as possible (less than 10 feet is preferred).

ESS-116: All RT isolation transformers shall be designed to provide an output signal level of 18 to 27V p-p at the component output interface.

3.4.1.2 SpaceWire Network Requirements

LRO-unique interface requirements for the high-speed instrument data are to be specified in the LRO Project SpaceWire Specification (431-SPEC-000103).

3.4.1.3 Ka-Band Downlink I&Q Data Interfaces

ESS-117: The Ka-band I-channel and Q-channel (I&Q) downlink data interface from the C&DH to the Ka-Band RF transmitter shall be fabricated per the Electrical Characteristics of Low Voltage Differential Signaling (LVDS) Interface Circuits (TIA/EIA-644).

3.4.1.4 Analog Telemetry Interfaces

ESS-118: The SC C&DH shall be capable of monitoring the passive analog, active analog and discrete telemetry signals.

ESS-119: The specific thermistor part number utilized to generate the passive analog telemetry shall be identified in the Thermal Systems Specification (431-SPEC-000091)

ESS-120: 5 kilohms bias resistor shall be used in parallel with the thermistor. The range of the temperature can be sensed by the thermistor with the bias resistor is -30 to +70 degree Centigrade (ºC) as represented by 0 - 5V telemetry output.

ESS-121: To prevent the current switching noise from the passive analog returns, the active analog signal ground or reference shall not be shared with the passive analog returns.

3.4.1.5 +5 V Discrete Command Interfaces

ESS-122: A +5 V discrete pulse command shall be used to actuate digital actuators.

ESS-123: A +5 V discrete pulse shall be transmitted using RS-422 differential signals

ESS-124: A +5 V discrete pulse shall adhere to the electrical terminations as given in the Electrical Characteristics of Balanced Voltage Digital Interface Circuits (TIA/EIA-422).

ESS-125: +5V discrete command receivers shall use a 100 ohm termination resistor to match the impedance of the RS-422 cable.

ESS-126: + 5V discrete command transmitters shall use 45 ohm inline resistors which together with the output impedance of the transmitter, match the impedance of the RS-422 cable. The required termination configuration is shown in Figure 3-12.

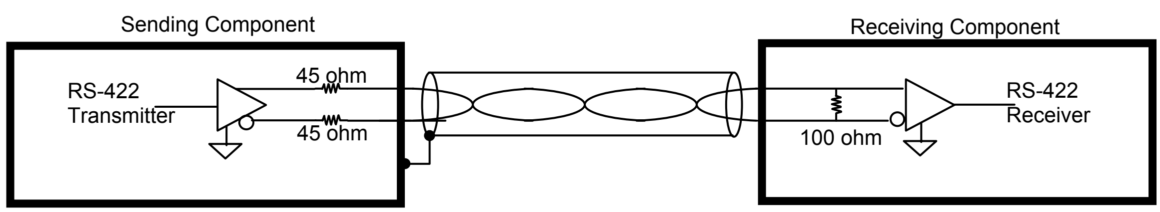


Figure 3-12. +5V Discrete Command termination

3.4.1.6 +28 V Discrete Command Interfaces

The +28V discrete pulse command will be used to actuate relays or other non-digital actuators.

ESS-127: The discrete command pulse circuit shall be designed to minimize coupling of the actuator switching noise into the digital or logic portion of the electronics.

ESS-128: Unregulated +28 VDC from the PSE may be used to generate the +28 V pulse command; however, the isolation requirements between primary power and secondary power (logic power) of 1Mohm shall always be maintained.

3.4.1.7 1 Pulse per Second (1 pps)

ESS-129: The LRO SC shall provide a 1 Pulse per Second (pps) to subsystems that require SC timekeeping information.

ESS-130: The 1 pps shall be transmitted using RS-422 differential signals and shall adhere to the electrical terminations as given in Section 3.4.1.5 above.

ESS-131: Users of the 1 pps shall use receivers and termination compatible with the C&DH which is using an Intersil HS-26CLV31RH differential driver.

ESS-132: The 1 pps shall have timing characteristics as shown in Figure 3-13 and Table 3-6 (below). The fiducial occurs on the rising edge of 1pps_int.

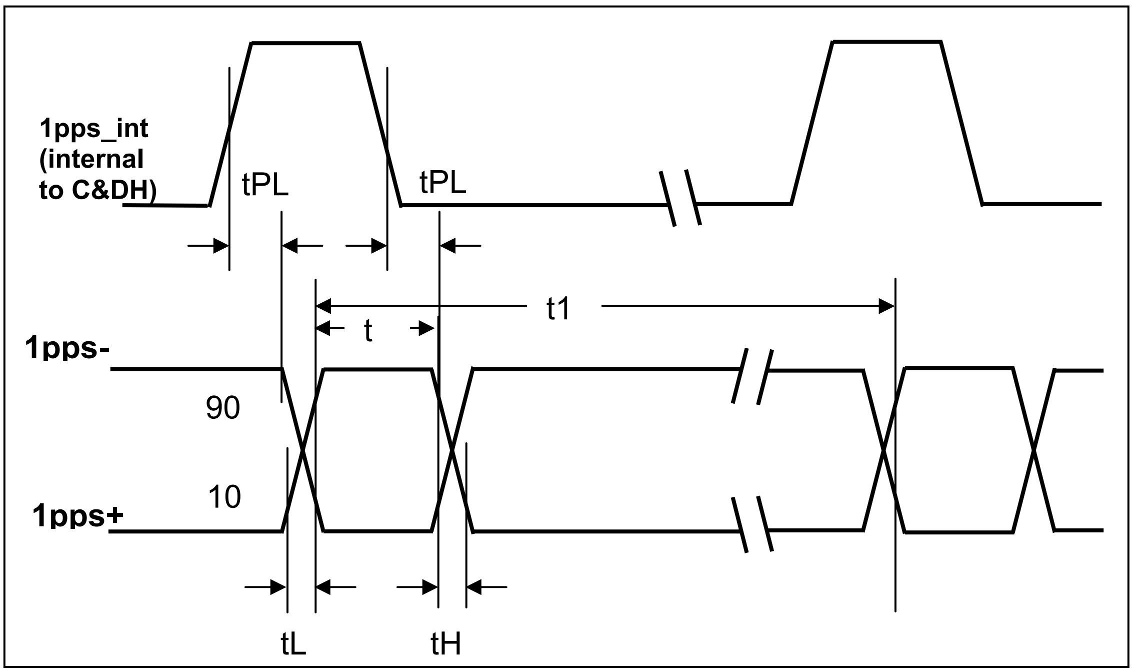


Figure 3-13. 1 pps Timing Characteristics

Table 3-6. 1 pps Timing Characteristics

Parameter	Value	Description
tPLH, tPHL	2-30 nanoseconds	Delay through driver
tTLH, tTHL	1-13 nanoseconds	Rise & Fall times
tH	34-36 microseconds	Pulse width
t1S	1s +/- 10 nanoseconds	Pulse-to-pulse accuracy

3.4.1.8 Other Inter-Component Communications

ESS-133: Other component interfaces not specified in the above Sections 3.4.1.1 through 3.4.1.5 shall be identified in the specific applicable subsystem to subsystem Electrical ICDs.

3.4.2 Pyrotechnic and Deployable Actuator Interfaces

ESS-134: The pyrotechnic and deployable control circuits utilized for the LRO pyrotechnic-actuated valves and deployable devices shall meet the Range Safety User Requirements (AFSCM 91-710).

The following LRO-unique requirements are derived from these mission success requirements:

ESS-135: The actuator initiation circuit shall have three independent inhibits, group enable, individual arm, and individual actuation (or fire) functions.

ESS-136: The group enable command, which are four separate enables, shall be utilized for enabling one of four actuator circuits: HGA deployment, SA deployment, pyrotechnic valve normally open (NO), and pyrotechnic valve normally closed (NC).

ESS-137: The pyrotechnic actuation circuits shall limit stray currents at the pyrotechnic device to less than 20 dB below the no-fire current level.

ESS-138: The actuation device and its control circuits shall be isolated from any external energy source by greater than 20 dB.

ESS-139: Electroexplosive devices shall be protected from electrostatic hazards by the placement of resistors from line-to-line and from line-to-ground (Structure). The parallel combination of all resistors must be 10k ohms or more (MIL-STD-1576).

ESS-238: The open/close position of the pyrotechnic valves shall be remotely monitored during pre-launch operations, or the functional power shall be deenergized (an additional fourth inhibit shall be in place between the power source and the three required inputs) and the control circuits for the three required inhibits shall be disabled (no single failure in the control circuitry will result in the removal of an inhibit) until the hazard potential of inadvertent activation no longer exists.

3.4.3 External Interfaces

ESS-140: LRO shall provide test points and other external interfaces to support the component integration, anomaly investigation, environmental testing, and launch site operations.

External electrical interfaces and access panels will be defined in the individual subsystems Electrical ICDs.

The LRO SC is to be launched on an expendable launch vehicle. Specific interface requirements between Orbiter GSE and the launch vehicle, and between the SC and the launch vehicle, will be defined in the applicable launch vehicle ICD(s).

3.4.3.1 Umbilical Interface Connector

ESS-141: The electrical interconnection between Orbiter and the launch vehicle shall be made through the umbilical connectors.

ESS-142: There shall be two umbilical interface connectors that are capable of carrying up to 37 each of 22 AWG or larger contacts to support the launch pad operations with the vehicle.

ESS-143: The umbilical connector(s) shall, at a minimum, include the following functions: hard-line control of the battery relay, battery charging/SA Simulator input and monitoring, hard-line communications, health-and-safety critical hard-line (H/L) telemetry data, and launch vehicle interface signals.

ESS-144: Umbilical signals shall be protected from ESD threats. Covering the umbilical connector(s) following the separation from the launch vehicle is difficult to achieve, therefore filtering is a preferred method of the ESD protection.

3.4.3.2 Launch Vehicle Separation Signals

ESS-145: Three independent separation switches shall be required to sense the valid separation from the launch vehicle.

These signals will be utilized to inhibit the SA deployment, high-gain antenna deployment, RF transmitters, thruster valves and the Reaction Wheel power on following the successful separation.

3.4.3.3 Component Test Interfaces

ESS-146: Component test points that require access during Orbiter-level testing shall be brought to the Orbiter skin connector

ESS-147: All test points shall be protected or isolated from the facility-induced noise, ESD, and GSE malfunction.

ESS-148: GSE cable connectors that mate with flight test connectors shall be flight-approved connectors.

ESS-149: Test connectors shall be capped with flight-approved RF and static control covers when not in use, including in orbit.

ESS-150: Wherever possible, component power shall not be applied or accessed at or through a test connector.

Test signals and flight signals should not be located together in the same connector.

3.5 MULIPACTION AND CORONA

ESS-151: Damage or measurable degradation due to RF breakdown (corona and arcing) shall be prevented by design.

ESS-152: The RF design shall preclude measurable degradation due to multipaction and corona in RF systems that must operate during the launch and ascent stages (e.g., filters, switches, and antenna elements), at critical pressures, or in a vacuum environment.

ESS-153: Components with high-voltage circuits shall be immune to corona and arcing while in a nominal orbital vacuum environment.

3.6 DESIGN FOR RADIATION

ESS-154: LRO components shall meet their performance requirements with acceptable degradation due to radiation induced effects. The LRO radiation requirements are contained within the Radiation Requirements for the Lunar Reconnaissance Orbiter (431-RQMT-000045).

3.7 CHARGING AND DISCHARGING REQUIREMENTS

ESS-155: The Orbiter shall meet all its performance requirements in the presence of various charging and discharging environment effects induced by the LRO trans-lunar and lunar orbit environments.

3.7.1 External Surface Charging

The Orbiter will be in a direct insertion lunar orbit, including one pass through the Van Allen belts, a high-charging plasma environment. It has been shown that in these areas, charge builds up and discharges, causing either a direct hit (lightning strike) or a possible large ground current that can disrupt the SC electronics. These can discharge between Orbiter elements (differential charging) or can discharge back to space (bulk charging).

3.7.1.1 Differential Charging

ESS-156: The Observatory shall minimize differential charging of the Orbiter surfaces.

ESS-157: The Orbiter, when exposed the photoelectron environment shall exhibit maximum differential charging less than the insulation breakdown voltage, with a factor of 2 margin.

3.7.1.2 Charging Mitigation

ESS-158: The Orbiter design shall prevent surface and internal charging/ discharging effects that can damage the SC components or disrupt SC operations.

ESS-159: External surfaces >6 cm^2 shall be conductive with a resistivity of less than 10^9 ohms per square (ohms/sq.) and grounded to the Observatory structure per Section 3.2.5, so that charge can bleed from that surface faster than the charge can build up on that surface.

ESS-160: There shall be no more than 10 surfaces of <6 cm^2 in size with surface resistivity greater than 10^9 ohms/sq. on the outer surface per 1 square meter of external surface.

ESS-161: Insulating films such as Kapton and other dielectric materials on the external surface shall be less than 5 mil thick and assembled to minimize surface charge build-up and grounded to bleed surface charge.

Non-conductive surface areas on the SAs and instrument optics will be minimized to the extent possible.

ESS-162: System impacts of the discharges from any unavoidable non-conductive surfaces shall be assessed and approved. If necessary, a waiver request will be submitted for each specific non-compliant application.

ESS-163: A comprehensive list of non-conductive surfaces on the Orbiter shall be maintained and reviewed periodically throughout the LRO development to identify any new ESD threats to the Orbiter.

ESS-164: The list shall include assessment of each dielectric region of the Orbiter surface for its breakdown voltage, its ability to store energy, and its effects on neighboring electronics (disruption or damage) and surfaces (erosion or contamination).

ESS-165: The Orbiter design shall minimize bulk charging of the Observatory relative to the space plasma.

ESS-166: Surfaces exposed to the Sun shall contain >1m^2 conductive surface that is grounded to Orbiter structure in order to bleed off accumulated charges back to space.

ESS-167: Proper handling, assembly, inspection, and test procedures shall be developed to insure the electrical continuity of the Orbiter surface grounding.

ESS-168: Surface conductivity shall be verified by measuring resistance from any one point of material surface to the Orbiter structure.

ESS-169: All grounding methods shall be demonstrated to be acceptable over the service life of the Orbiter.

3.7.2 Surface Discharging Protection

ESS-170: The Orbiter shall be designed to prevent discharges on the external surfaces from permanently damaging components or upsetting science data collection.

ESS-171: The electrical system shall be designed to carry discharge currents and to shield from the electric field from the discharge without any permanent damage to the Orbiter.

ESS-172: SA panels shall use materials and fabrication techniques to minimize ESD effects.

ESS-173: Power system electrical design shall incorporate features to protect against transients due to electrical discharge from the SA.

ESS-174: SC transmitters and receivers (command line and data line) shall be immune to transients produced by ESD.

ESS-175: The transmitter, receiver, and antenna system shall be tested for immunity to ESD near the antenna feed.

ESS-176: The repetition rate shall be selected to be consistent with estimated arc rates of nearby materials.

ESS-177: The Orbiter structure shall be designed to serve as a Faraday cage providing 40 dB overall shielding of the internal electronics from the external environment:

- The Faraday cage consists of the K1100 structure panels or an RF shield providing 40 dB attenuation for any aperture not covered with panels.
- An RF shield can be the equivalent of at least four layers of VDA2 material, with each layer grounded to chassis along its entire perimeter.
- Joints, seams, and seals between panels can be constructed from copper tape with conductive adhesive or EMI gaskets. Gaps should be less than 2.5 cm in length.

ESS-178: The Faraday cage shall not have any gaps or holes larger or longer than 2.5 cm required to maintain 40 dB shielding to 500 MHz from the external noisy environment.

ESS-179: All internal harnesses and electronics shall be kept 20 cm away from uncovered or shielded structure openings or apertures that will be present in flight.

ESS-180: There shall be no openings that provide a direct path for charged plasma to pass into the Observatory structure (Faraday cage).

ESS-181: The Orbiter electrical elements and harnesses outside of the Orbiter mechanical bus structure (Faraday cage) shall be protected from the discharges generated by ESD sources:

ESS-182: Sufficient shielding shall be used around cables to protect against circuit damage or operational upsets from discharges and their RF emissions.

A minimum 1 mil aluminum foil or aluminum tape wrapped with a 50% overlap around external harnesses up to the entrance of the Faraday cage can be used to seal off the external environments. It should be noted that additional shielding will be necessary to prevent the internal charging due to the Teflon insulation of the harness (see Section 3.7.3).

ESS-183: External circuits shall be filtered at entrance to the Faraday cage. If filtering is not practical at the entrance to the Faraday cage, the harness shield shall be continued internal to the Faraday cage after being sealed and grounded at the entrance, and the harness wires shall be shielded and separately routed to their destination component and then filtered at the component.

ESS-184: External sensors and harness (i.e., thermistors) shall have an outer shield that is grounded 360 degrees circumferentially at the entrance to the Faraday cage.

ESS-185: Shields terminated at the entrance to the Faraday cage shall not protrude more than 2.5 cm into the Faraday cage.

ESS-186: Inner shields connected to the outer shield shall not be brought through a connector pin into the Faraday cage.

ESS-187: Exposed voltages above 50V shall be protected from shorts to ground caused by a plasma cloud generated by a discharge.

External harness charge mitigation is represented pictorially in Figures 3-3 (above) and 3-14 (below).

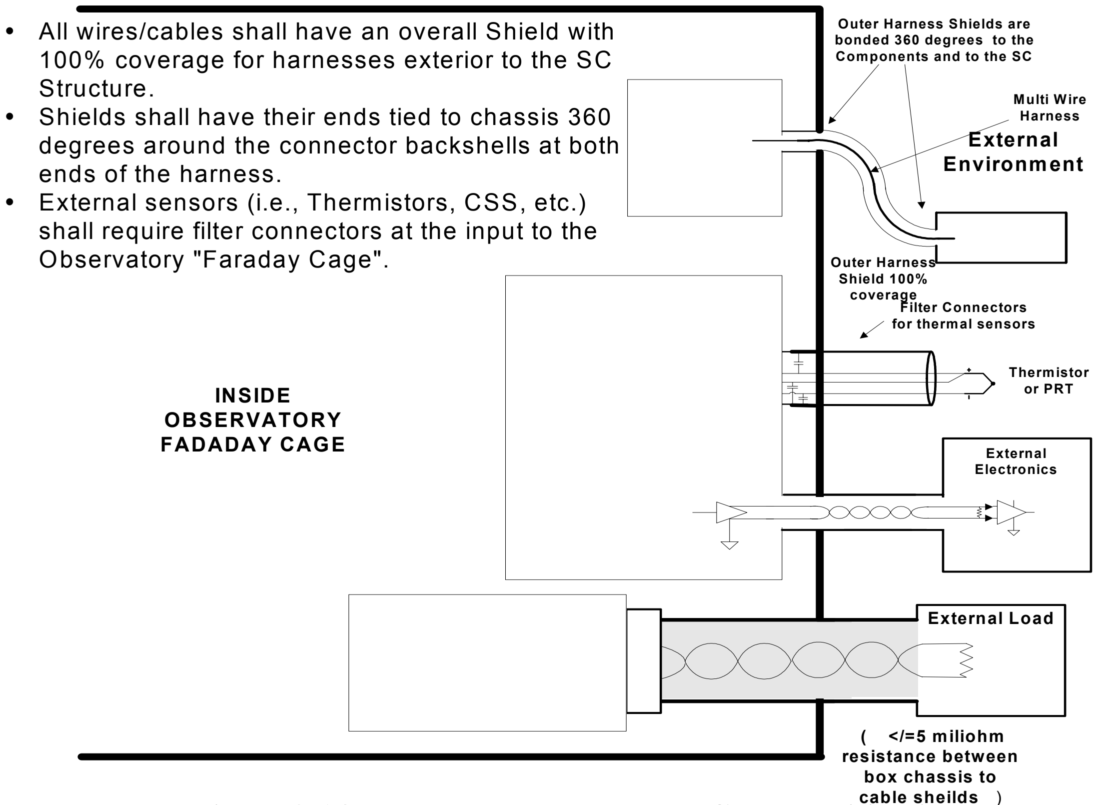


Figure 3-14. External Harnesses and Charge Mitigation

3.7.3 <u>Internal Charging</u>

ESS-188: The Orbiter design shall prevent internal charging/discharging effects that may damage the Observatory components or disrupt Orbiter operations. Internal dielectrics and metal charging/discharges occur from those energetic electrons that penetrate the Orbiter surfaces.

ESS-189: Internal charging effects shall be controlled by shielding all electronics elements with sufficient aluminum equivalent thickness (100 mil Al for bulk dielectrics or to 60 mil Al equivalent for Teflon harness insulation) so that the internal charging rate is benign.

ESS-190: Internal dielectrics materials with bulk resistivity of $>10^{12}$ ohm-cm that do not meet the shielding requirement shall be controlled via one of the methods described in the following paragraphs below. Materials typically include unavoidable dielectrics such as the SA insulators, antennas, connectors, thermal isolators, thermistor mounting, thermal blankets, thermal blanket tape, and other materials such as Kapton and Teflon insulators, which have a bulk resistivity of higher than 10^{18} ohm-cm. The following techniques are to be used:

- Limit the electron flux to insulators by shielding to 10^{10} electrons/cm^2 in 10 hours. (This can be met with plate shielding with 110 mil Al for bulk dielectrics or to 60 mil Al equivalent for Teflon harness insulation.)

- Filter nearby circuitry to withstand a 5,000-V, 20-pico Farad (pF), 10-ohm discharge.

- Detailed analysis of discharge could result in smaller or larger discharge source than above. Assess discharge threat to circuits that cannot be totally shielded such as antennas, umbilical connector, thermistors, platinum resistance temperature (PRT) sensors, heaters, SA, Coarse Sun Sensor (CSS), etc.

- Coat the exterior surface of the dielectric with a grounded layer with a resistivity of $<10^9$ ohm/sq.

- Prevent the discharge from reaching a victim circuit by EMI shielding and or grounded conductive barrier that will safely absorb and dissipate the discharge.

ESS-191: If none of above control techniques can be applied, the impacts of the discharge from the dielectrics material shall be assessed for an approval.

ESS-192: Ungrounded (floating) conductors shall not be allowed in the Orbiter. This includes unused wires in harnesses; ground test sensors; ground use signals in umbilical cables; unused or unpopulated circuit board traces; ungrounded integrated circuit (IC), relay, transistor, or capacitor cases; spare pins in connectors; thermal blankets; aluminum or copper tape; ungrounded bracketry for harness or connectors; TC105 harness tie-down clips; harness P-clamps; conductive epoxy; thermostat cases; screws; or nut plates. Exceptions are allowed by waiver if analysis shows that no direct or radiated path to victim circuitry exists or that the victim can survive discharge.

ESS-193: Leakage impedance of conductive internal parts shall be less than 10,000 ohms. This requirement applies to conductive fittings on dielectric structural parts. Further investigation into these effects and mitigations of internal charging can be found in the Avoiding Problems Caused by Spacecraft On-Orbit Internal Charging Effects Handbook (NASA-HDBK-4002).

4.0 HARNESS REQUIREMENTS

ESS-194: Orbiter harnesses shall satisfy the requirements of this section and the electrical systems requirements described in Section 3 of this specification.

ESS-195: Harnesses shall be developed Design and Development Guidelines for Spaceflight Electrical Harnesses (565-PG-8700.2.1).

ESS-196: Qualified wire, cable, and connector specified in the Instructions for EEE Parts Selection, Screening, Qualification, and Derating (EEE-INST-002) shall be used for the SC flight harness or any of the component harnesses.

ESS-197: Wires, connectors, connector contacts, and other harness piece parts shall be derated per the Instructions for EEE Parts Selection, Screening, Qualification, and Derating (EEE-INST-002).

ESS-198: Minimum wire sizes shall be per Table 4-1, subject to the limitations inherent in the Power System Electronics (PSE) connectors.

Figure 4-1. Minimum and Maximum AWG

Type	Maximum AWG (minimum size)	Minimum AWG (maximum size)
2 Amp Unswitched Services	22	22
5 Amp Unswitched Services	22	20
1 Amp Switched Services	22	22
2 Amp Switched Services	22	20 (limit 3)
5 Amp Switched Services	22	20
10 Amp Switched Services	20	20
15 Amp Switched Services	20	20
Heater Circuits	22	22
SpaceWire	28	26
Signals (other than SpaceWire)	24	24

4.1 GENERAL HARNESS GUIDELINES

Harnesses should be designed to meet the functional requirements of the subsystems, and the requirements given in Section 3.0, while adhering to the following guidelines:

ESS-199: Wherever possible, the harnesses shall be grouped by common electrical characteristics of the signals carried in the harness.

ESS-200: All power lines and power return lines to a particular component shall be twisted together in a single bundle. Power harnesses should be routed away from signal harnessing, and power and signal wires should not be run in the same bundle.

ESS-201: Wherever two or more power conductors are used to increase the current carrying capability, or provide redundant wires, the component interfaces to the two conductors shall be of the same design

ESS-202: Wherever two or more power conductors are used to increase the current carrying capability, or provide redundant wires, the wires shall be routed in the same harness to keep path length the same for both

ESS-203: Wherever two or more power conductors are used to increase the current carrying capability, or provide redundant wires, there shall be the same number of power and return lines, and each return wires shall be twisted with a power wire

ESS-204: Wherever possible, the power and signal shall not share the same component interface and harness connectors.

ESS-205: The connector half that sources power to another component shall be female (socketed) to protect against inadvertent grounding prior to mating.

ESS-206: The pyrotechnic harness shall be a twisted, shielded cable without any interruptions in the outer shield.

4.1.1 Accessibility

Harnesses should be designed with accessibility and manufacturability in mind:

ESS-207: The component connectors, on each box, shall be spaced far enough apart to access the harness connector with EMI backshells by a hand or with an extraction tool.

a) Any harness cable or connector should not touch any other adjacent connectors or harnesses.

b) Electrical boxes should be spaced 6 inches (15 cm) from a panel or structure mounting strut when the box has one or more connectors that face that structure.

c) Electrical boxes should be spaced 6 inches (15 cm) apart when one of these boxes has connectors that face the other box.

d) Electrical boxes should be spaced 9 inches (23 cm) apart when both of these boxes have connectors that face each other.

ESS-208: Mechanical support for harnesses shall be designed in accordance with the Crimping, Interconnecting Cables, Harness, and Wiring (NASA-STD-8739.4).

ESS-209: Harness splices shall not be allowed without LRO project approval.

ESS-210: Wires that have tin coating shall not be used for flight due to the possibility of tin whisker growth that could cause a short. Silver is the preferred coating.

4.1.2 Harness Shields

ESS-211: Harness shields shall be terminated at the connector backshell at both ends of each harness:

ESS-212: All shields shall be grounded to the SC structure.

ESS-213: Wire harness shields shall not carry current by design.

ESS-214: Outer shields shall not be tied to component connector pins or wires in the cable bundle.

Inner-shield pig-tail lengths to ground should be minimized, and should not exceed 5 cm.

ESS-215: Shield termination pig-tails shall be bonded to the connector backshell.

Pig-tails should only pass through a connector pin when used as a special noise reduction technique (as in the case of LVDS [or SpaceWire Cable] inner shields).

ESS-216: Shields shall not be daisy-chained from one shield to another. Each internal harness shielded wire should connect its shield directly to the ground point.

ESS-217: EMI backshells shall be required for all harness connectors on the external surfaces of the Orbiter.

ESS-218: All external harnesses shall be shielded with an overall outer bundle shield and terminated with 360° outer shield bond to the backshell.

ESS-219: Aluminum tape (LG-1055 or equivalent) at least 1 mil thick, wrapped with 50% overlap on each 360-degree wrap over the previous wrap shall be used as harness bundle shield. For the external harnesses, the additional shielding may be required to protect from the deep dielectric charging.

ESS-220: All internal interconnections and cables between components shall be bundle shielded and terminated.

4.1.3 Component Test Connector Panels

ESS-221: Component test connector panels shall be located on an external Orbiter surface, and these panels should be accessible during various I&T and launch pad operations.

ESS-222: All electrical connectors shall be covered such that they are not exposed to the trans-lunar and lunar orbit environments.

4.1.3.1 Safing Plugs or Arming Plugs

ESS-223: Safing and arming plugs shall be incorporated in the cable or harness that control ordinance, deployable actuators, propulsion valves, RF transmitters, SA power, Battery power, electromechanical actuator devices and lasers.

4.1.3.2 Fairing Access to Connector Plugs

ESS-224: Access through the launch vehicle fairing shall be provided for contingency operations such as fuel off-loading, safety related inhibit/critical circuit arm plug removal, contamination event inspection, long term storage/maintenance and limited troubleshooting.

The emphasis is on completing all required SC/payload access requirements prior to fairing installation to simplify pad operations, access through fairing doors should be not be planned for routine pad operations.

4.1.3.3 Fuse Pigtails

ESS-225: Power harnesses shall use break-out-boxes (BOB) for fuses to permit the interruption of the unswitched power bus to a faulty component or harness during I&T.

4.2 ELECTRICAL MATERIALS

ESS-226: All subsystem materials list shall be reviewed for their electrical properties and assessed for its compatibility with electrical systems requirement defined in this specification.

ESS-227: Parts with unstable materials (in terms of the LRO orbital environment) that cannot be stabilized through additional processing for the proposed application shall not be used.

4.2.1 Connectors

ESS-228: All connectors on a component shall be different sizes, pin counts and/or genders to prevent any mismating of harness to component connectors.

ESS-229: Wherever possible, keying shall be used.

ESS-230: Connector savers shall be used during integration and test to minimize wear on connector contacts.

ESS-231: Connector mate/demate logs shall be used to record mates and demates.

ESS-232: Instrument developers are expected to provide the mating connectors to their flight components for use on the SC harness.

ESS-233: All subsystems shall provide a list of connectors to the SC electrical systems engineer prior to connector part procurement.

ESS-234: All component input/output (I/O) connector interfaces shall be designed to accommodate an EMI backshell.

5.0 QUALIFICATION ASSURANCE PROVISIONS

5.1 GENERAL

All requirements in this document shall be verified by one of the four methods defined below.

5.1.1 Analysis

The analysis method is used when:

- A rigorous, representative, and conclusive analysis is possible
- Test is not cost effective, and
- Inspection and demonstrations are not adequate

Analyses may include, but are not limited to, engineering analysis (which includes models and simulations), review of record, and similarity analysis.

5.1.1.1 Engineering Analysis

Engineering analysis may be quantitative, qualitative, or a combination of the two. Quantitative analysis involves the study and modeling of the physical entity whose performance is to be verified. Examples of quantitative analyses include end-to-end link analysis, structural (static and dynamic) analysis, thermal models, pointing knowledge and stability. Qualitative analyses are non-numerical and related to qualitative measure of performance, such as failure modes and effects analyses (FMEA), maintainability, and redundancy.

5.1.1.2 Validation of Records and Other Documentation Analysis

This kind of analysis uses design and manufacturing documentation to show compliance of design features and manufacturing processes. Validation of design documentation, e.g., engineering drawings, verifies that the “as-designed” hardware complies with contractual design and construction requirements. Validation of manufacturing records at end-item acceptance verifies that the “as-built” hardware has been fabricated per the approved design and associated documentation. Review and analysis of other documentation such as acceptance data packages and other compliance documentation of lower levels of assembly are valid analysis techniques.

5.1.1.3 Similarity Analysis

Similarity is included as a valid verification/qualification method. Qualification by similarity is used in lieu of test when it can be shown that an item is similar to, or identical in design to another item that has been previously qualified to equivalent, or more stringent requirements. Formal qualification documentation of the previously qualified item must be available for assessment when planning to qualify by similarity. Furthermore, an item whose design has been qualified by similarity must undergo acceptance verification to assess workmanship.

5.1.2 Demonstration

Demonstration is a verification method that provides a qualitative determination, rather than direct quantitative measurement, of the properties or functional characteristics of an end-item. The qualitative determination is made through observation with, or without test equipment or instrumentation.

5.1.3 Inspection

Inspection is the verification method used to verify construction features, workmanship, dimension, physical characteristics, and spacecraft conditions such as configuration, cleanliness, and locking hardware. Inspection also includes simple measurements such as length, and it is performed without the use of special laboratory or precision equipment. In general, requirements specifying function or performance are not verified by inspection.

5.1.4 Test

Verification by test consists of direct measurement of performance parameters relative to functional, electrical, mechanical, and environmental requirements. These measurements are obtained, during or after controlled application of functional and environmental stimuli to the test article, e.g., payload or satellite, and using instrumentation or special test equipment that is not an integral part of the test article being verified. The test activities include reduction and analysis of the test data, as appropriate. The following paragraphs define different categories of tests including performance, functional, environmental, interface, and structural tests.

5.1.4.1 Performance Test

A performance test consists of an individual test or series of electrical and/or mechanical tests conducted on flight, or flight-configured hardware and software at conditions equal to, or less than design specifications. Its purpose is to verify compliance of the test article with the stated applicable specification requirements that are verifiable by test. Typically, a full performance test is conducted at ambient conditions at the beginning and the end of a test sequence during which the test article is subjected to applicable environmental conditions, e.g., vacuum, high/low temperature extremes, or acoustics/random mechanical excitation.

5.1.4.2 Functional Tests

A functional test is a suitably chosen subset of a performance test. Typically, functional tests are conducted at ambient conditions between environmental exposures during the qualification or acceptance test sequence. The objective is to verify that prior to application of the next environment, exposure to the environment has not adversely affected the test article. When appropriate, functional tests, or a portion thereof, are conducted while the test article is exposed to a particular thermal or vacuum environment. Functional test, or a portion thereof, may also be conducted to assess the state of health of the hardware after major operations, such as transportation of flight hardware from one location to another.

5.1.4.3 Environmental Tests

Environmental testing is an individual test or series of tests conducted on flight, or flight-configured hardware to assure that flight hardware will perform satisfactorily after it is subjected to the induced launch environments, as well as its flight environment. Examples are: vibration, acoustic, temperature cycling, thermal vacuum and vacuum outgassing certification, and Electromagnetic Interference/Compatibility. Depending on the severity of the chosen environmental conditions, the purpose of the environmental exposure is to sufficiently stress the hardware so as to verify the adequacy of the design (protoflight levels and durations) or workmanship during fabrication (acceptance levels and durations).

5.1.4.4 Special Tests

Special tests are individual tests, or a series of tests conducted on flight, or flight-configured hardware to assure satisfactory performance of a particular critical element of the system, e.g., optical alignment. The special test verification category includes structural, mechanism and communication tests. Special tests may, or may not be performed in conjunction with environmental exposure.

5.1.4.5 Interface Tests

Interface tests verify the mechanical, electrical, and/or hardware-software interface between units and elements integrated into a higher level of assembly such as a module, subsystem, element, or a system.

5.1.4.6 Structural Tests

These tests are performed on structural elements, components, or assembled subsystems before delivery of the assembled structure to the integration and test organization. Structural tests designed to verify requirements of this specification may include: (1) static structural proof tests (to verify the strength/stiffness adequacy of the primary load path), and (2) dynamic tests, such as a modal survey or acoustic response test.

5.2 VERIFICATION MATRIX TABLE

The following matrix table defines the method of verification for all requirements contain in this document:

Table 5-1. Verification Matrix Table

Verification Method:		Level:
Inspection	(I)	II System
Analysis	(A)	III Segment
Demonstration	(D)	IV Element
Test	(T)	V Subsystem

Requirement Number	Section Number	Object Heading	I	A	D	T	Responsible Org.
TBD							

Appendix A. Abbreviations and Acronyms

Abbreviation/ Acronym	DEFINITION
A/µs	Amps per microsecond(s)
AC	Alternating Current
AFSPCMAN	Air Force Space Command Manual
Al	Aluminum
Am^2	Amp meters squared
ANSI	American National Standards Institute
AWG	American Wire Gauge
BC	Bus Controller
BOB	Breakout Box
C	Centigrade
C&DH	Command and Data Handling
CE	Conducted Emissions
CCAS	Cape Canaveral Air Station
CCB	Configuration Control Board
CCD	Charge-Coupled Device
CCR	Configuration Change Request
CM	Configuration Management
cm^2	Centimeters squared
CMO	Configuration Management Office
CRaTER	Cosmic Ray Telescope for the Effects of Radiation
CS	Conducted Susceptibility
CSS	Coarse Sun Sensor
dB	Decibel
dBm	Decibel meter
dBµA	Decibel microamps
DC	Direct Current
DLRE	Diviner Lunar Radiometer Experiment
ECSS	European Cooperation for Space Standardization
EEE	Electrical, Electronic, Electromechanical
EIA	Electronic Industries Alliance
ELV	Expendable Launch Vehicle
EMC	Electromagnetic Compatibility
EMI	Electromagnetic Interference
ESA	European Space Agency
ESD	Electrostatic Discharge
ETU	Engineering Test Unit
GEVS	General Environmental Verification Standard for GSFC Flight Programs and Projects
GHz	Gigahertz

Abbreviation/ Acronym	DEFINITION
GSE	Ground Support Equipment
GSFC	Goddard Space Flight Center
H/L	Hard-line
HDBK	Handbook
HGA	High Gain Antenna
Hz	Hertz
I&Q	I-channel and Q-channel
I&T	Integration and Test
I/O	Input/Output
IC	Integrated Circuit
ICD	Interface Control Document
IMU	Inertial Measurement Unit
INST	Instruction
IRW	Integrated Reaction Wheel
LEND	Lunar Exploration Neutron Detector
LOLA	Lunar Orbiting Laser Altimeter
LRO	Lunar Reconnaissance Orbiter
LVDS	Low Voltage Differential Signaling
kHz	kilohertz
kohm	kilohms
krad	kilorads
μV/m	Microvolts per meter
μs	Microsecond(s)
mA/μs	milliamps per microseconds
Mbps	Megabits per second
MDM	Micro-D Metal
MHz	megahertz
MIL	Military
m^2	Meter squared
mm^2	Millimeters squared
Mohm	Megohms
ms	millisecond
mV	millivolts
NASA	National Aeronautics and Space Administration
NHB	NASA Handbook
NC	Normally Closed
NO	Normally Open
ohms/sq.	Ohms per square
p-p	peak-to-peak
PCB	Printed Circuit Board

Abbreviation/ Acronym	DEFINITION
PDE	Propulsion / Deployment Electronics
pF	pico Farad
PRT	Platinum resistance temperature
PSE	Power Subsystems Electronics
RF	Radio Frequency
RLEP	Robotic Lunar Exploration Program
rms	Root Mean Squared
RQMT	Requirement
RS	Radiated Susceptibility
RT	Remote Terminal
SA	Solar Array
SBC	Single Board Computer
SC	Spacecraft
SEE	Single-Event Effects
SEECA	Single-Event Effect Criticality Analysis
SEU	Single-Event Upset
SPG	Single-Point Ground
SPEC	Specification
SSPC	Solid-State Power Control
ST	Star Tracker
STD	Standard
STS	Space Transportation System
T/M	Telemetry
TBD	To Be Determined
TBR	To Be Resolved
TIA	Telecommunications Industry Association
V	Volt(s)
V/m	Volts per meter
VDC	Volts Direct Current
Vpp	Volts peak-to-peak

Appendix B. Traceability Matrix

Parent Requirement			Requirement			Child Requirement		
RQMT#	**Section#**	**Object Heading**	**RQMT#**	**Section#**	**Object Heading**	**RQMT#**	**Section#**	**Object Heading**
TBD								

NOTE: Each Requirement must have its own Object Heading.